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R-ASTR-S-69-7  
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# TECHNICAL INFORMATION SUMMARY

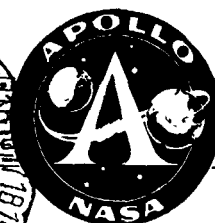
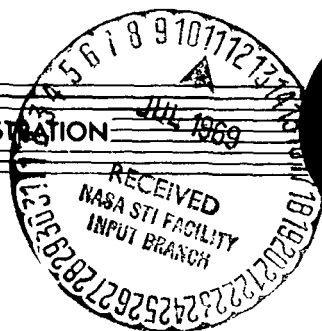
## APOLLO-9 (AS-504)

APOLLO SATURN V  
SPACE VEHICLE

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## AS-504

### TECHNICAL INFORMATION

#### SUMMARY

This document is prepared jointly by the Marshall Space Flight Center Laboratories R-ASTR-S, R-AERO-P, and R-P&VE-VN. The document presents a brief and concise description of the AS-504 Apollo Saturn Space Vehicle. Where necessary, for clarification, additional related information has been included.

It is not the intent of this document to completely define the Space Vehicle or its systems and subsystems in detail. The information presented herein, by text and sketches, describes launch preparation activities, launch facilities, and the space vehicle. This information permits the reader to follow the space vehicle sequence of events beginning a few hours prior to liftoff to its journey into space.

#### 1. Mission Purpose:

Mission D will be flown to perform CSM/LM operations in earth orbit, particularly to evaluate lunar module system capabilities, to include LM rendezvous technique and the combined CSM/LM functions.

#### 2. Mission Objectives:

AS-504 will be a manned space flight with an S-IVB second burn (high ellipse) and a third S-IVB burn to achieve earth escape.

##### The principal test objectives are:

a. CSM active docking, LM extraction and verification of S-IVB attitude control.

b. Complete lunar module checkout to include; landing gear deployment, landing radar, long duration Ascent Propulsion System burn, Descent Propulsion System duration, Lunar Module Environmental Control System performance, LM docking, LM Electrical Power Subsystem, LM structural integrity, Extra Vehicular Activities and LM ejection from Spacecraft Lunar-Module Adapter.

##### The secondary test objectives are:

a. Determine capability of S-band, high gain antenna to acquire, track and communicate with MSFN.

b. Verify one-man LM operational capabilities, obtain data on crew activities.

### 3. Mission Description

Mission D, AS-504 has been designed for a ten day duration.

The Space Vehicle Mission is divided into six phases: (1) launch, (2) parking orbit, (3) pre-ignition sequencing and second burn, (4) intermediate orbit coast, (5) pre-ignition sequencing and third burn, and (6) final coast.

Launch Phase. The Apollo Saturn V Space Vehicle, (AS-504), will be launched from Complex 39 pad A, Kennedy Space Center, and will be rolled to a flight azimuth of 72 degrees east of true north. As the vehicle rises from the launch pad, a yaw maneuver is executed to insure that the vehicle does not collide with the tower in the event of high winds or engine failure. Once tower clearance has been accomplished, a tilt and roll maneuver is initiated to achieve proper flight attitude and flight azimuth orientation.

Boost to Earth Parking Orbit is accomplished as illustrated in Figure 1. The S-IVB/IU, LM CSM will be inserted into a 103 nautical mile Earth Parking Orbit at approximately 10 minutes and 58 seconds ground elapsed time (G.E.T.) after liftoff. This insertion requires a complete burn of the S-IC and S-II stages and a partial S-IVB burn. During this phase, the flight performance of the S-IC stage will be monitored to observe the modifications which have been incorporated to suppress low frequency, launch vehicle, longitudinal oscillations.

Parking Orbit Phase. Following insertion into Parking Orbit, the vehicle will be commanded to maintain a cutoff attitude during which time a maneuver will be completed which will align the longitudinal axis parallel to the local horizontal and to establish and maintain an earth orbital rate. After approximately 2 hours and 34 minutes from liftoff, the space vehicle will be maneuvered to an inertial attitude which is suitable for transposition, docking, and extraction.

At approximately 2 hours and 40 minutes GET, using the CSM Reaction Control System (RCS) for maneuvering, the CSM will separate from the S-IVB/IU LM and the SLA panels will be jettisoned. Within a few seconds from separation, the CSM will be pitched 180 degrees and rolled 60 degrees so that it is positioned facing the S-IVB/LM. At approximately three hours GET, the CSM will complete a hard dock with the S-IVB/IU/LM while the vehicle is passing over western United States. Shortly after the space vehicle enters daylight, (4 hours - 10 minutes GET), the lunar module (LM) extraction is begun. The LM attach points are severed pyrotechnically thereby activating the spring thrusters which will give the CSM/LM a separation velocity. As soon as the CSM/LM is clear of the S-IVB/IU, the CSM RCS is used to move the CSM/LM away from the S-IVB/IU.

The command service module will prepare for the first Service Propulsion System (SPS) burn immediately following lunar module extraction. This maneuver is scheduled to take place while the space vehicle is over Hawaii at about 6 hours GET.

Pre-Ignition Sequencing and Second S-IVB Burn. After the CSM/LM separation an inhibit will be removed from the S-IVB/IU by ground command, allowing pre-ignition sequencing and the second S-IVB burn to be accomplished. During this phase of the mission, particular attention will be given to monitoring the operation of the  $O_2 - H_2$  burner (helium heater). During pre-ignition sequencing, the  $O_2 - H_2$  burner will be used as the primary repressurization system for the fuel tanks. The ambient repressurization system, used in previous Saturn V missions (AS-501 and 502) will be used as a backup in the event of  $O_2 - H_2$  burner malfunction. The  $O_2 - H_2$  burner uses LOX and  $LH_2$  to heat the helium supplied by the cold helium spheres, which are located in the  $LH_2$  tank. The heated helium gas expands and pressurizes the LOX and  $LH_2$  tanks prior to restart. Using the ambient system as the backup will require a longer pre-ignition sequencing. The second burn will last approximately 62 seconds.

Pre-Ignition Sequencing and Third S-IVB Burn. At approximately 6 hours and 7 minutes GET, following removal of an inhibit by ground command, the S-IVB will ignite for the Third burn of approximately 5 minutes duration. Prime consideration during the pre-ignition sequence will be to demonstrate the  $O_2 - H_2$  restart capability. The ambient system for repressurization will also be used. The  $O_2 - H_2$  burner will be ignited for about a 2 minute period.

Final Coast. When the S-IVB Engine is cutoff for the third time, a maneuver will be initiated to align the S-IVB longitudinal axis parallel with the local horizontal. Passivation will begin at approximately 6 hrs 12 minutes GET. The vehicle will be maneuvered to a position in order to accomplish the following; LOX and  $LH_2$  dump thru the J-2 engine, open the propulsive and non-propulsive vents in the LOX and  $LH_2$  tanks, and to open the cold helium valve. These events will require approximately 63 minutes for completion.

#### Spacecraft Mission Summary

Since the Apollo Mission D has been defined as a combined command service module and lunar module operations mission, the following is a summary of the mission D highlights to briefly emphasize the principal events and the approximate time of their occurrence.

(See figures 1 and 2)

#### EVENT

APPROXIMATE  
Ground Elapsed Time (GET)  
(DAY: HR: MIN)

#### Phase I

- Boost to Earth Orbit (S-IVB/CSM) - - - - -	~ 0:00:11
- S-IVB/CSM Separation (SLA Panel Jettison) - - - -	~ 0:02:39
- CSM/LM Docking and LM Extraction - - - - -	~ 0:02:53
- S-IVB 2nd Burn (High Ellipse) - - - - -	~ 0:04:45

## EVENT

APPROXIMATE  
Ground Elapsed Time (GET)  
(DAY: HR: MIN)

### Phase I (continued)

- Docked (CSM/LM) SPS 1st burn - - - - - ~ 0:06:00
- S-IVB 3rd burn (Escape) - - - - - ~ 0:06:07

### Phase II

- Docked (CSM/LM) 2nd., 3rd., 4th. SPS burn: 2nd -- 0:23:44  
3rd -- 1:03:14  
4th -- 1:06:20

### Phase III

- Docked burn of the LM Decent Propulsion (DPS). - - - - - 2:06:35

### Phase IV

- Astronaut Extra Vehicular Activity (EVA). Two of the 3-man crew will move in tethered mode from CSM to LM. - - - - - 2:19:00
- Crew will operate a 5th SPS burn of the CSM for undocking. - - - - - 3:10:12

### Phase V

- LM undocks from CSM (LM is manned) - - - - - 3:21:00
- LM staging will separate the Ascent from the Decent Portion of the LM - - - - - 4:06:52
- Following an undocked APS burn, the CSM and LM will rendezvous and the crew will return to the CSM. - 4:09:18

### Period VI

- LM will jettison from the CSM - - - - - 4:21:00
- APS (LM) Final Burn (to move jettisoned LM out of plane) - - - - - 4:22:19
- 6th SPS burn of CSM - - - - - 4:27:30

EVENT

APPROXIMATE  
Ground Elapsed Time (GET)  
(DAY: HR: MIN)

Period VI

- 7th SPS burn of CSM (to establish favorable  
deorbit conditions) - - - - - 5:06:23
- From the 5th to the 9th day, the crew activities  
consist primarily of navigation sightings, inertial  
measurements, S/C equipment usage and rest periods.
- At ~ 9 days and 19 hours the crew will begin preparation  
for re-entry.
- SPS deorbit burn of CSM - - - - - 9:21:46
- SM jettisoned from CM - - - - - 9:22:36
- CM re-entry - - - - - 9:35:00

Parachute descent and splashdown will follow re-entry.  
Approximately 10 days from lift off, the Command  
Module will splash down in the Atlantic at a point  
1000 NMI East of Cape Kennedy.

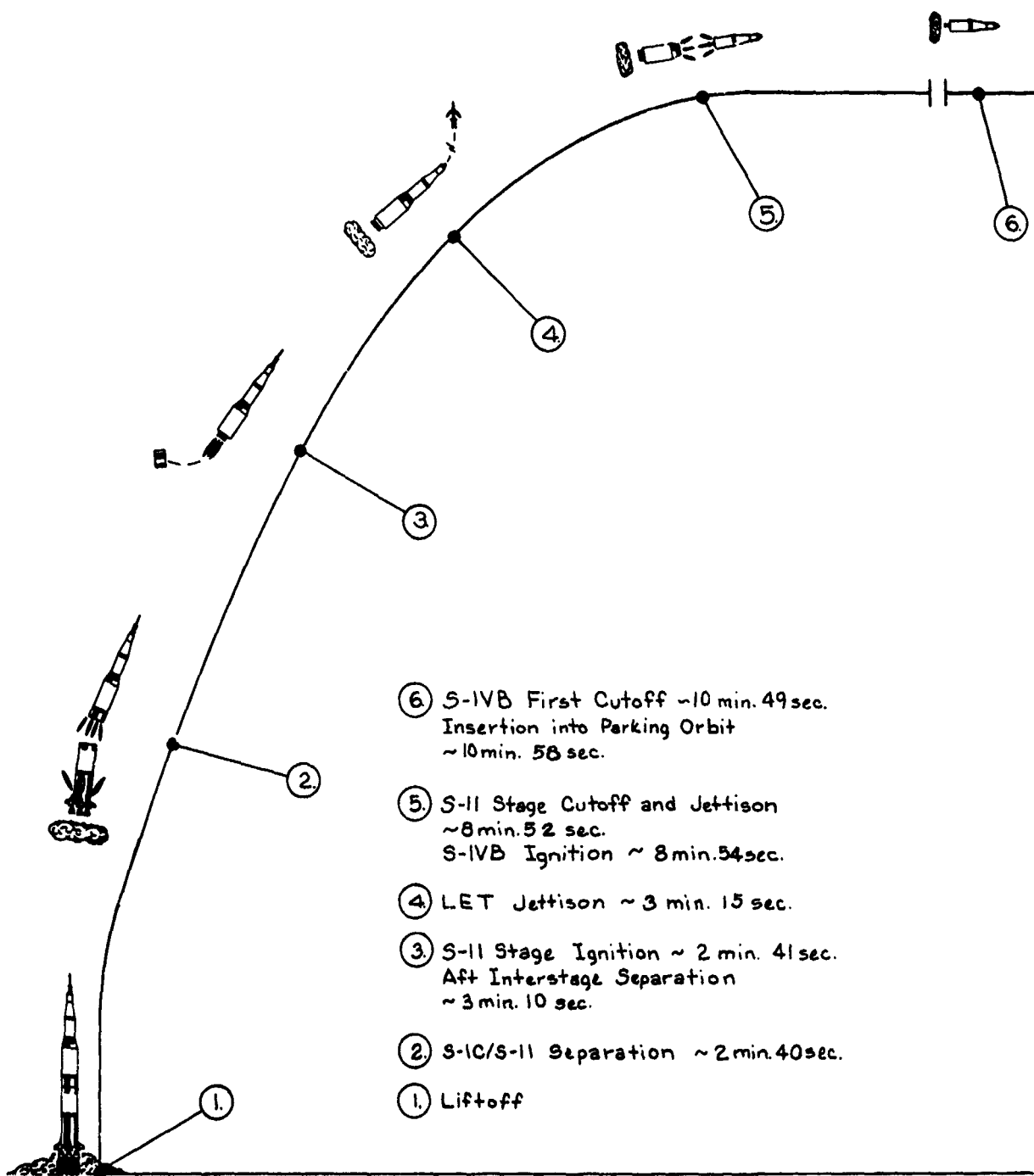


Figure 1

Mission Profile  
Boost to Earth Orbit

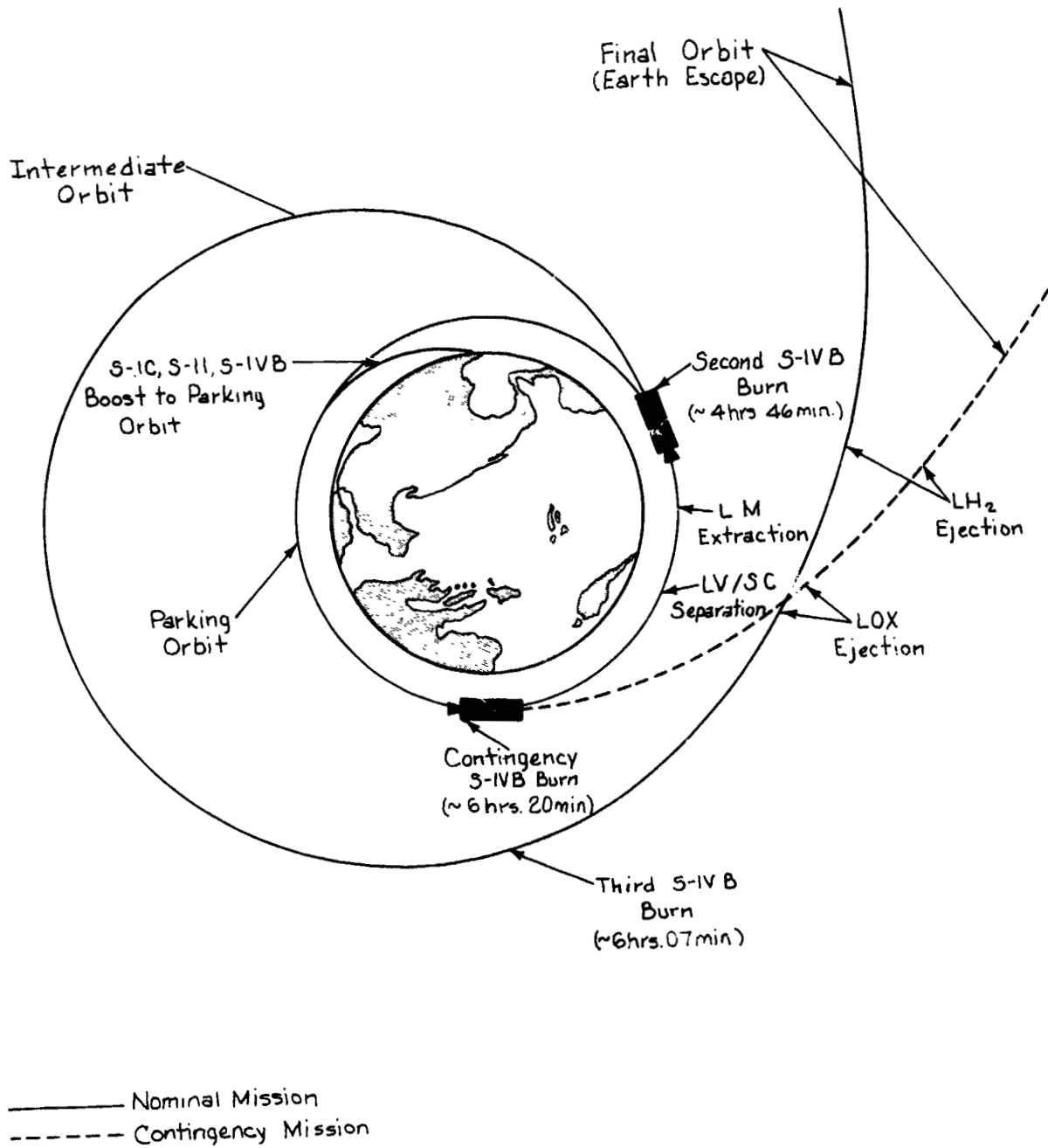


Figure 2

Mission Profile  
Space Vehicle Trajectory



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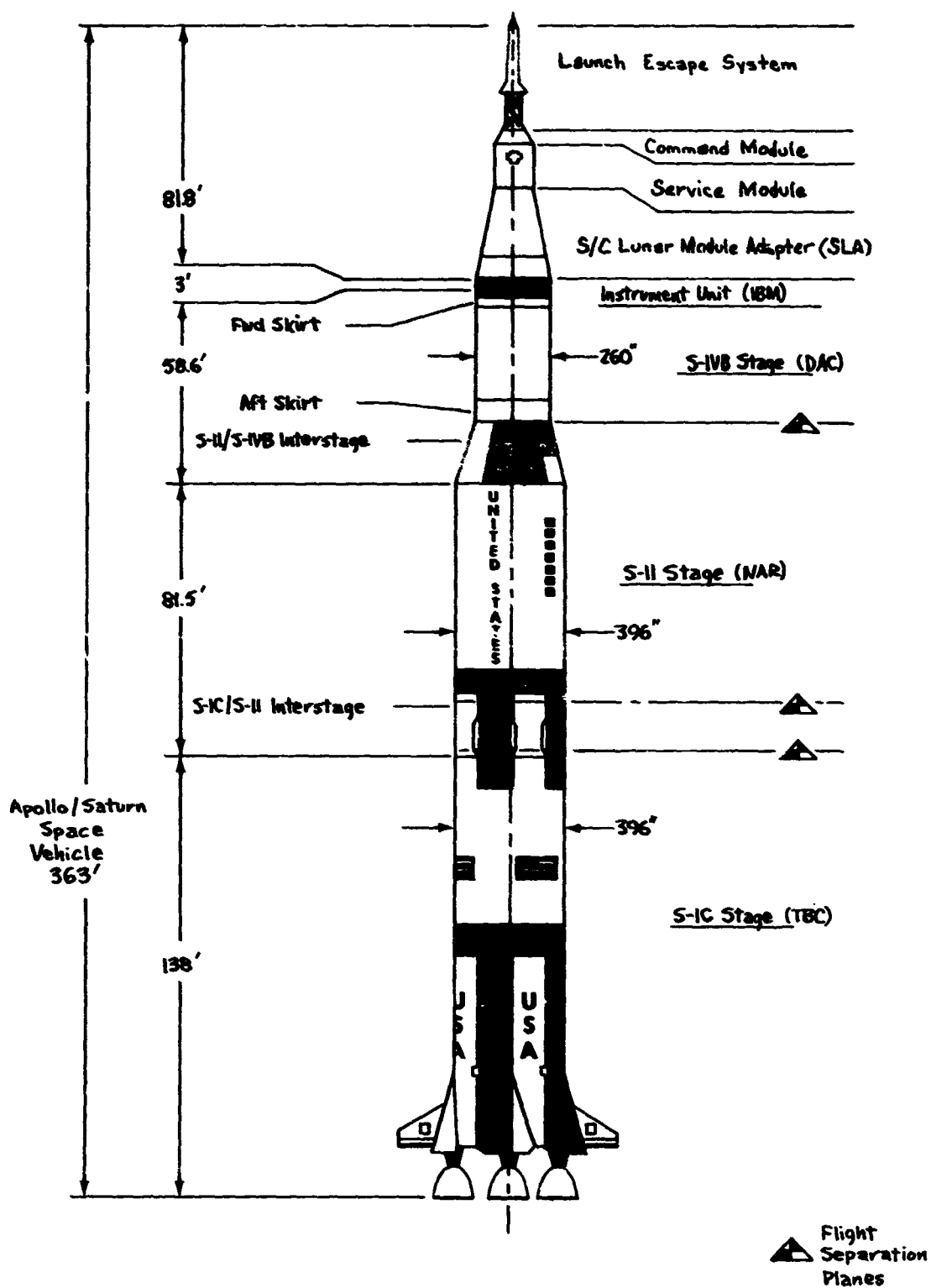
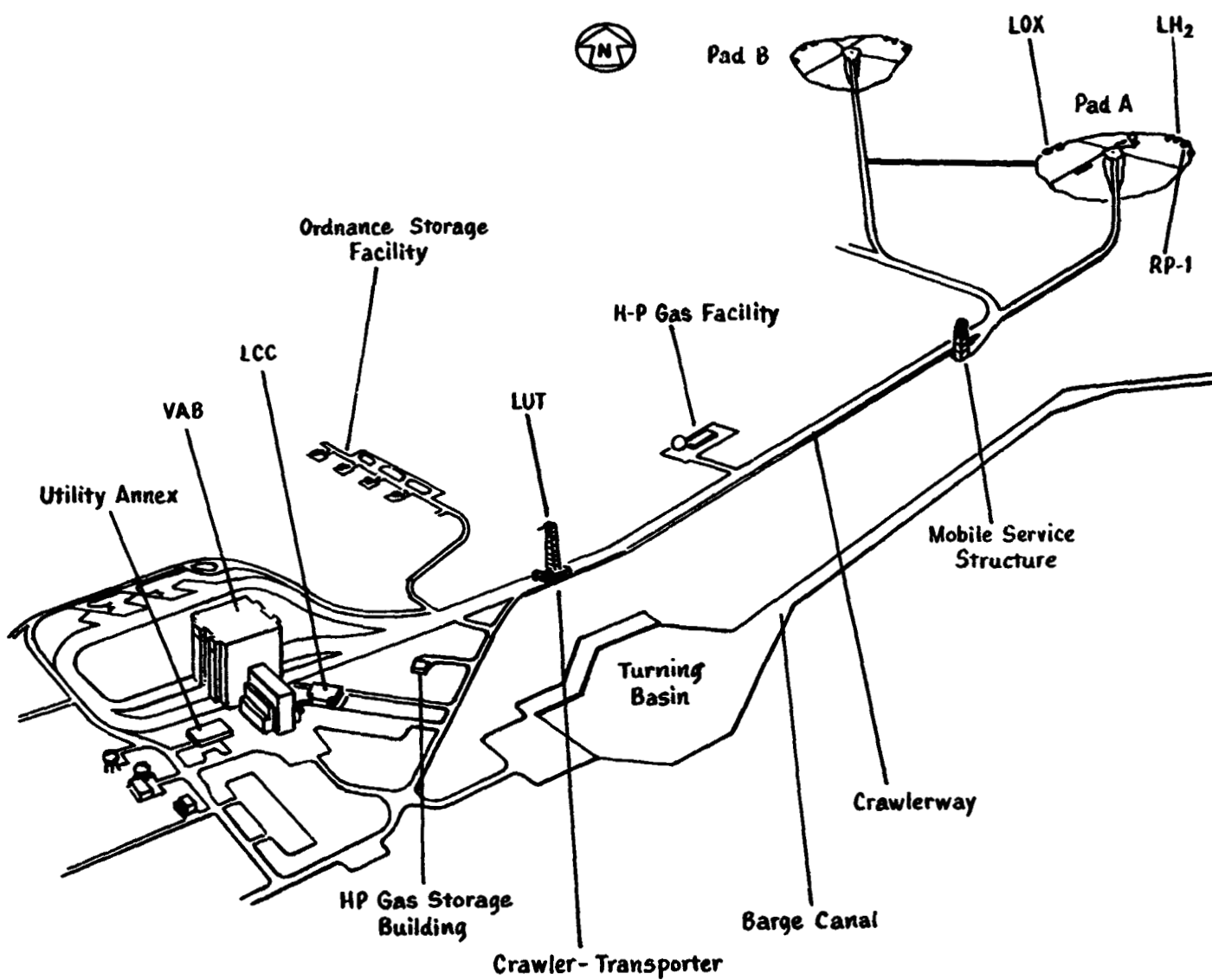


Figure 3

AS-504  
Space Vehicle



**Figure 4**

**Launch Complex 39**

## MOBILE LAUNCHER

The Mobile Launcher, figure 5, is a transportable steel structure which provides the capability of moving the erected vehicle to the launch pad via the crawler-transporter. The umbilical tower, permanently erected on the mobile launcher base, is a means of ready access to all important levels of the vehicle during assembly, checkout and servicing prior to launch. The intricate vehicle-to-ground interfaces are established and checked out within the protected environment of the Vertical Assembly Building (VAB) and then moved undisturbed aboard the mobile launcher to the launch pad. The description of each mobile service arm (numbered arrows below) correspond to the numbered arrows on the Mobile Launcher illustrated on the opposite page (figure 5).

- ① S-1C Intertank (preflight). Provides LOX fill and drain. Arm may be reconnected to vehicle from LCC. Retract time 8 seconds. Re-connect time ~5 minutes.
- ② S-1C Forward (preflight). Provides pneumatic, electrical, and air conditioning interfaces. Retracted at T-16.2 seconds. Retract time 8 seconds.
- ③ S-11 Aft (preflight). Provides access to vehicle. Retracted prior to liftoff as required.
- ④ S-11 Intermediate (inflight). Provides LH<sub>2</sub> and LOX transfer, vent line, pneumatic, instrument cooling, electrical, and air-conditioning interface. Retract time 6.4 seconds.
- ⑤ S-11 Forward (inflight). Provides GH<sub>2</sub> vent, electrical, and pneumatic interfaces. Retract time 7.4 seconds.
- ⑥ S-1VB Forward (inflight). Provides LH<sub>2</sub> and LOX transfer, electrical, pneumatic, and air-conditioning interfaces. Retract time 7.7 seconds.
- ⑦ S-1VB Forward (inflight). Provides fuel tank vent, electrical, pneumatic, air-conditioning, and preflight conditioning interfaces. Retract time 8.4 seconds.
- ⑧ Service Module (inflight). Provides air-conditioning, vent line, coolant, electrical, and pneumatic interfaces. Retract time 9.0 seconds.
- ⑨ Command Module Access Arm (preflight). Provide Access to spacecraft through environmental chamber. Arm controlled from LCC. Retracted 12° park position until T-4 minutes.

### Note:

Preflight arms are retracted and locked against umbilical tower prior to launch.

Inflight arms retract at vehicle liftoff on command from service arm control switches (located in hold-down arms).

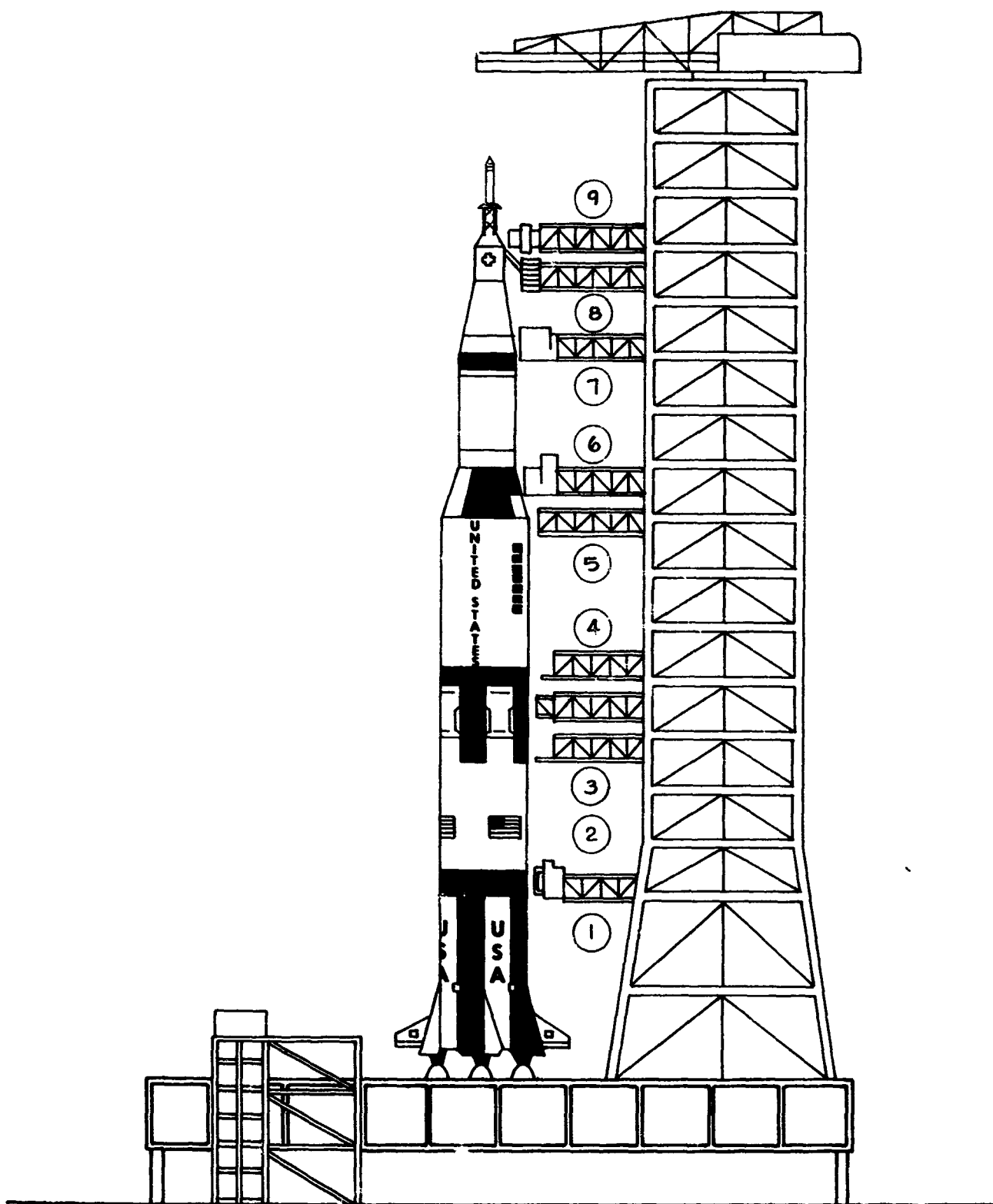


Figure 5

Saturn V  
Mobile Launcher

## LAUNCH VEHICLE SECURE RANGE SAFETY SYSTEMS

The Secure Range Safety Systems are located on the S-IC, S-II and S-IVB stages and are designed to provide a communication link for the transmission of coded commands from ground stations to the vehicle during boost phase. This transmission provides a positive means of terminating the flight of an erratic vehicle by initiating emergency engine cutoff and, if necessary, propellant dispersion.

The flight termination system in each powered stage consists of a range safety antenna subsystem, two secure command receivers, two Range Safety Controllers, two Secure Range Safety Decoders, two Exploding Bridge Wire (EBW) firing units, two EBW detonators and a common safe and arm device which connects the subsystem to the tank cutting charge. Electrical power for all elements appearing in duplicate is supplied from separate stage batteries.

Prior to launch, the safe and arm device is set to the "ARM" position by ground support equipment in the block house.

The S-IVB stage range safety receiver is commanded to an "OFF" condition by ground command at orbital insertion such that no destruct can take place after this safing action.

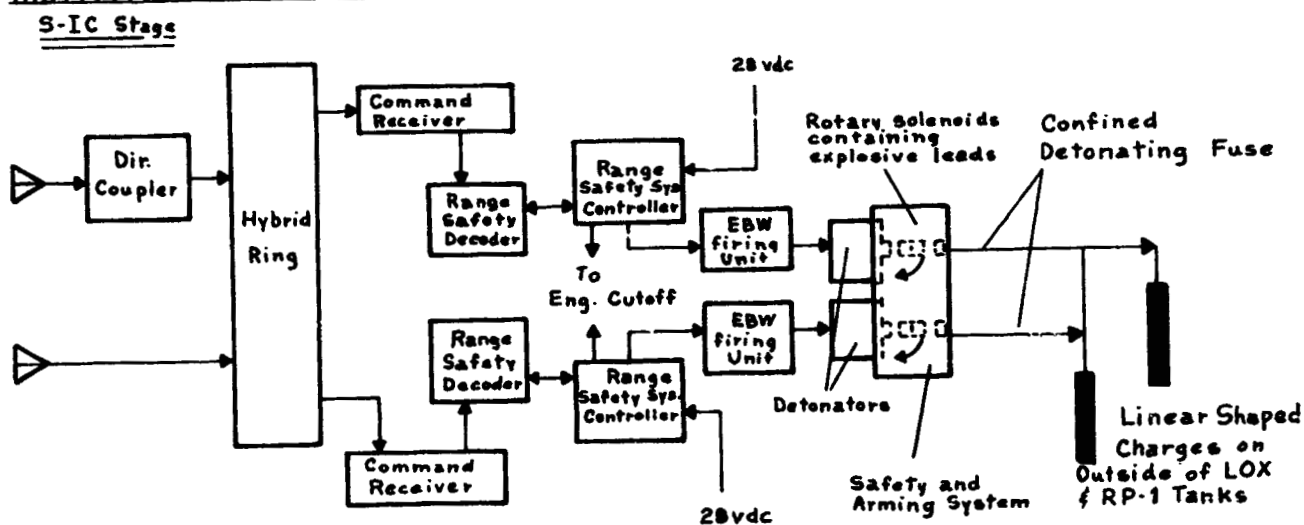
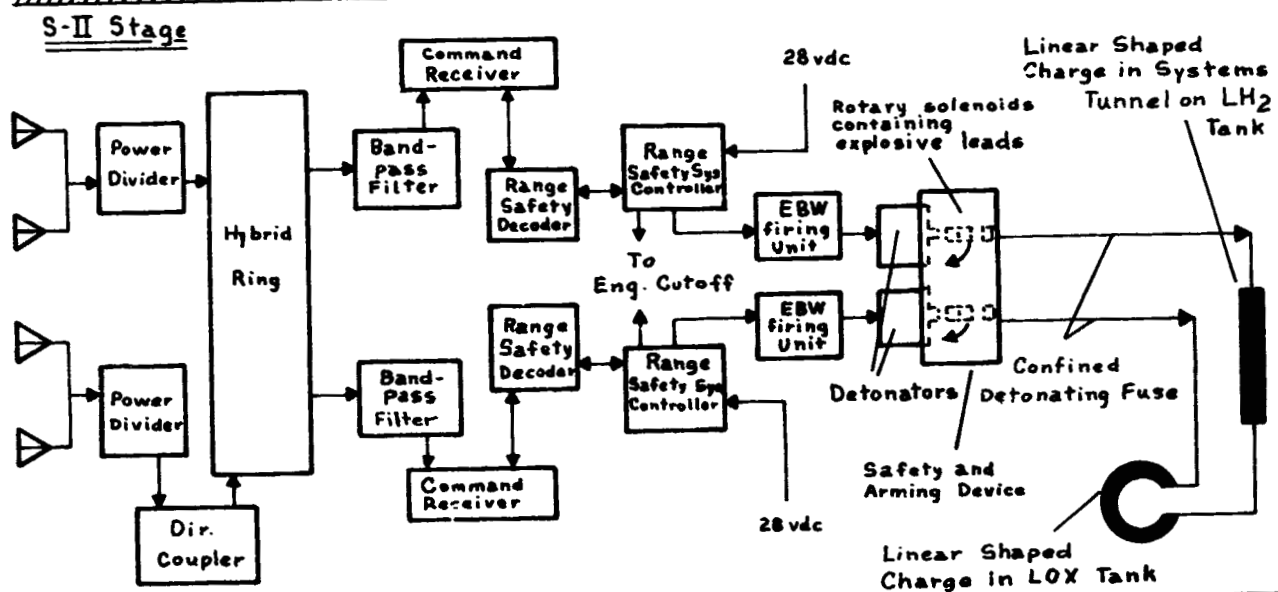
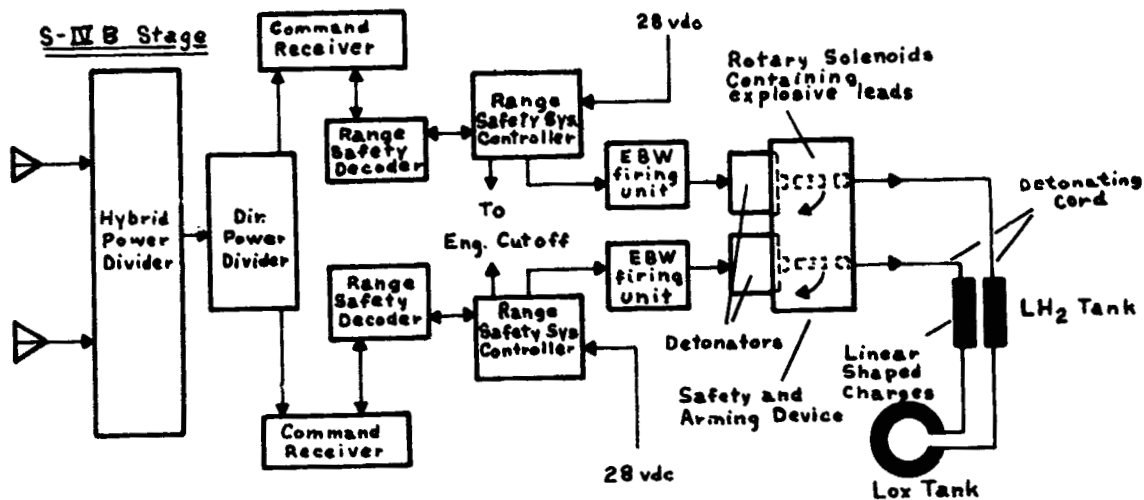


Figure 6

Secure Range Safety System



## EMERGENCY DETECTION SYSTEM (EDS)

The Emergency Detection System (EDS), which is a part of the Crew Safety System, is designed to sense and react to emergency situations resulting from launch vehicle malfunctions which may arise during the mission. Protection of the Apollo Crew against vehicle failure is the prime function of the EDS.

In general, the abort modes for operation of the EDS are:

Manual Abort - Based on Astronaut's judgment and decision.

Automatic Abort is initiated by excessive angular rates of the vehicle or by the loss of thrust in two or more engines in the S-IC stage during specified times of flight. The measurements are obtained from triple redundant sensors with majority voting logic.

The automatic abort rate limits are:  $\pm 4$  degrees per second with a tolerance of  $\pm .49$  degrees in pitch and yaw and  $\pm 20$  degrees per second with a tolerance of  $\pm 1.5$  degrees in roll.

Auto abort is automatically enabled at liftoff, provided the EDS auto, LV rates auto and 2 engine out auto switches are enabled in the spacecraft.

The automatic abort mode is active only during first stage flight from liftoff until the crew manually inhibits the automatic abort at approximately 120 seconds; therefore an automatic abort always utilizes the LES for escape (the LES is jettisoned shortly after S-II ignition by the crew).

In order to afford protection for personnel and facilities in the launch area, thrust is not terminated with aborts prior to 30 seconds of flight time. The switch selector enables the EDS cutoff circuitry at 30 seconds of flight with a timer backup also at 30 seconds.

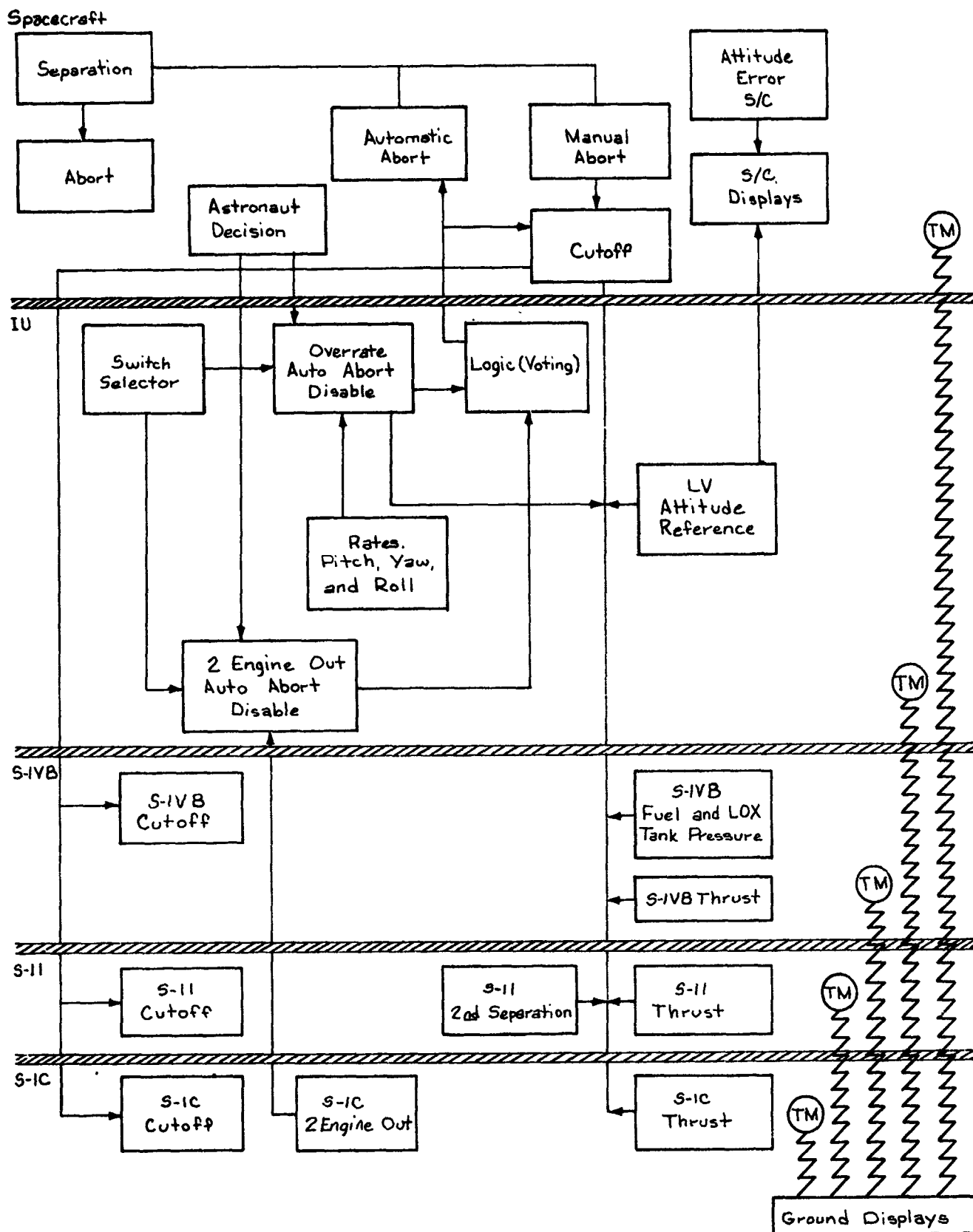


Figure 7

Emergency Detection System

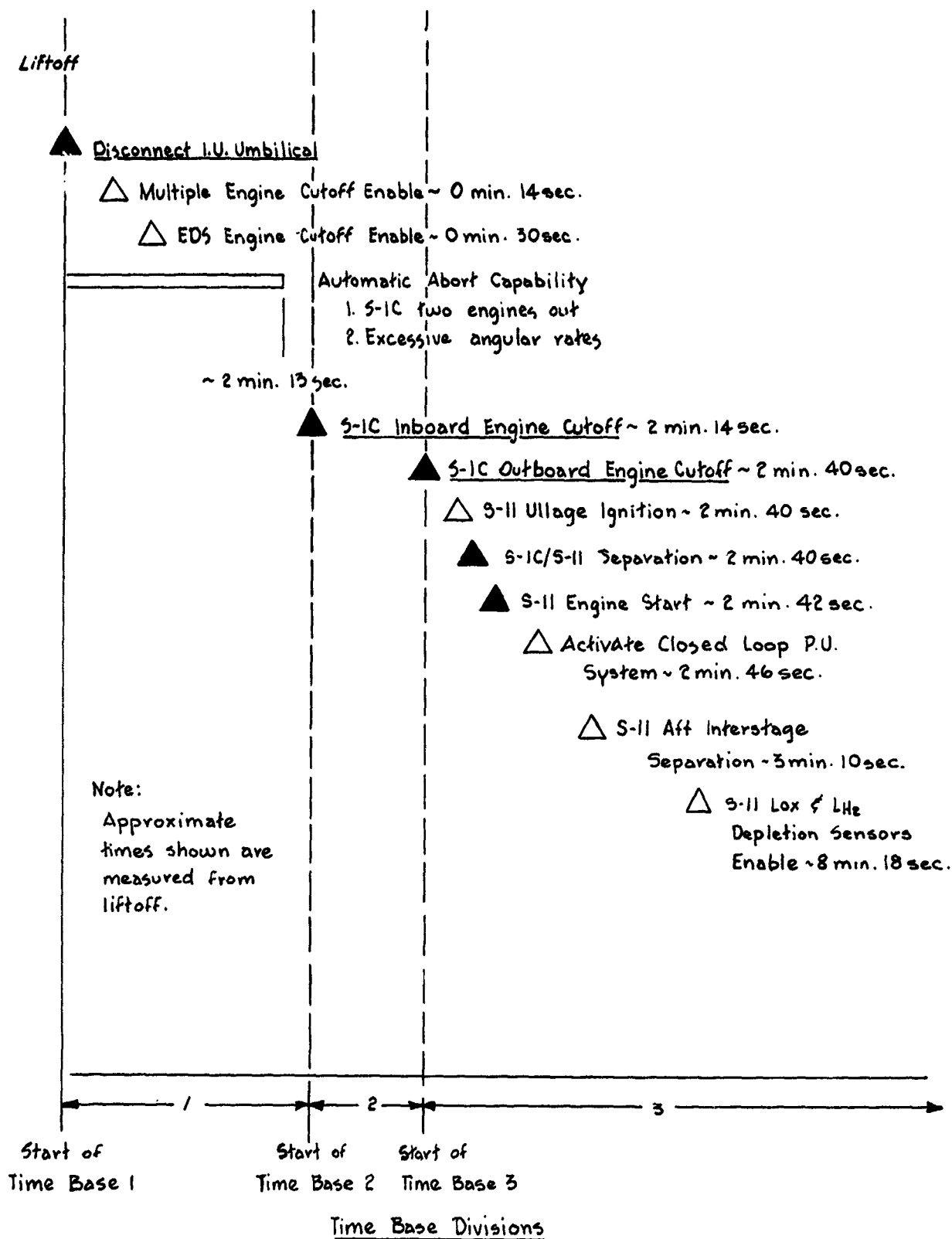


Figure 8

S-IC/S-II Stage  
Flight Sequencing

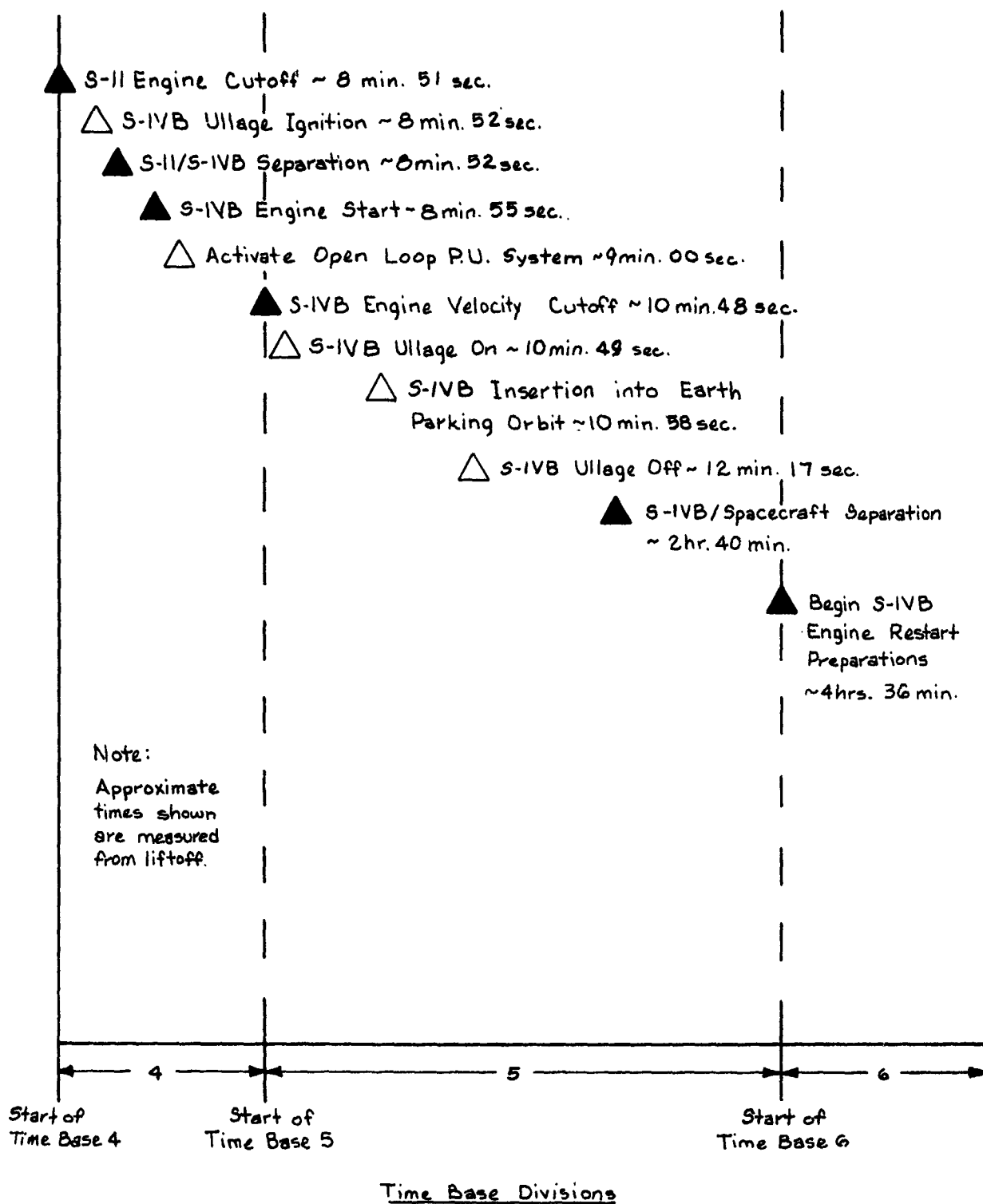


Figure 9

S-II/S-IVB  
Flight Sequencing

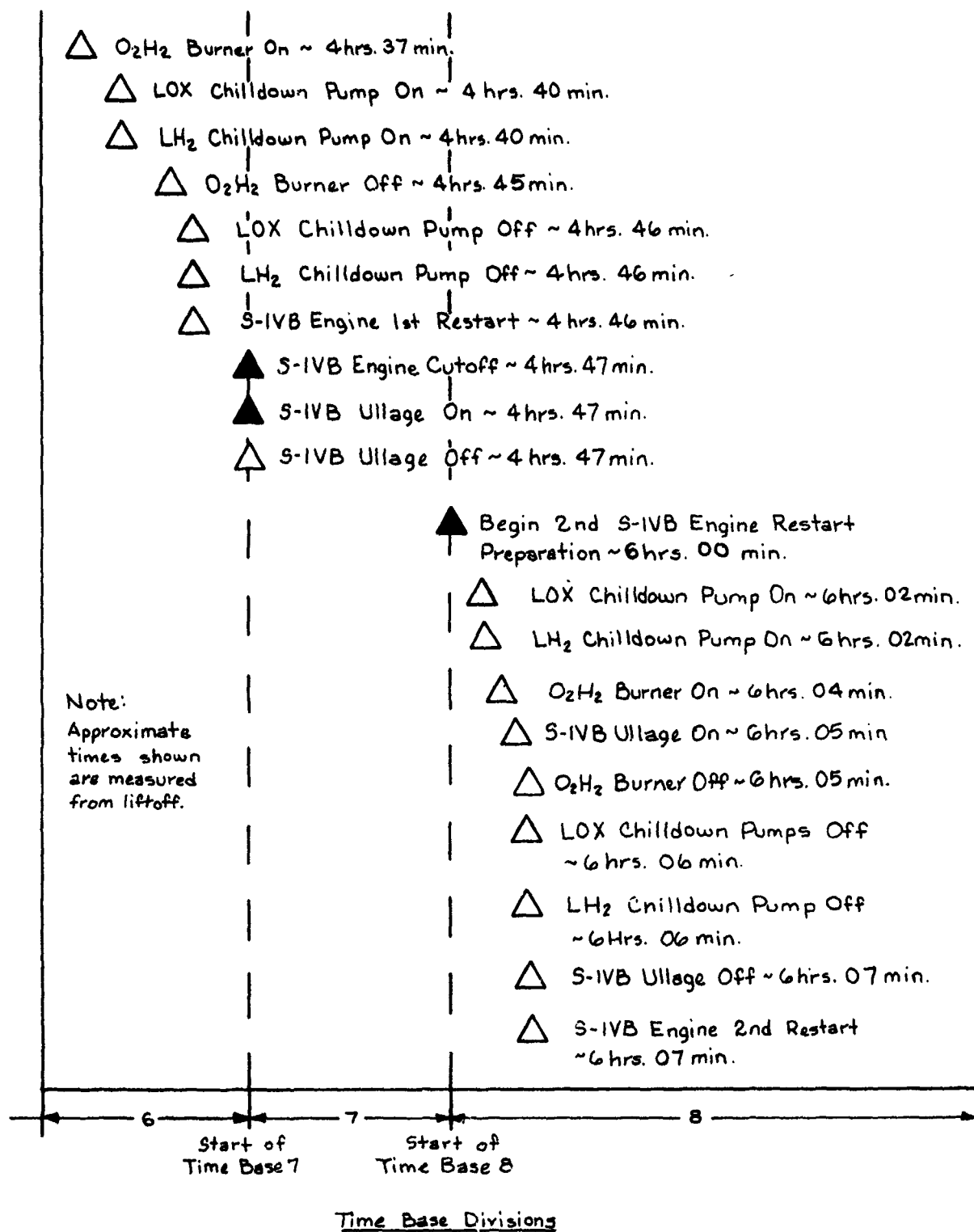


Figure 10

S-IVB Stage  
Flight Sequencing

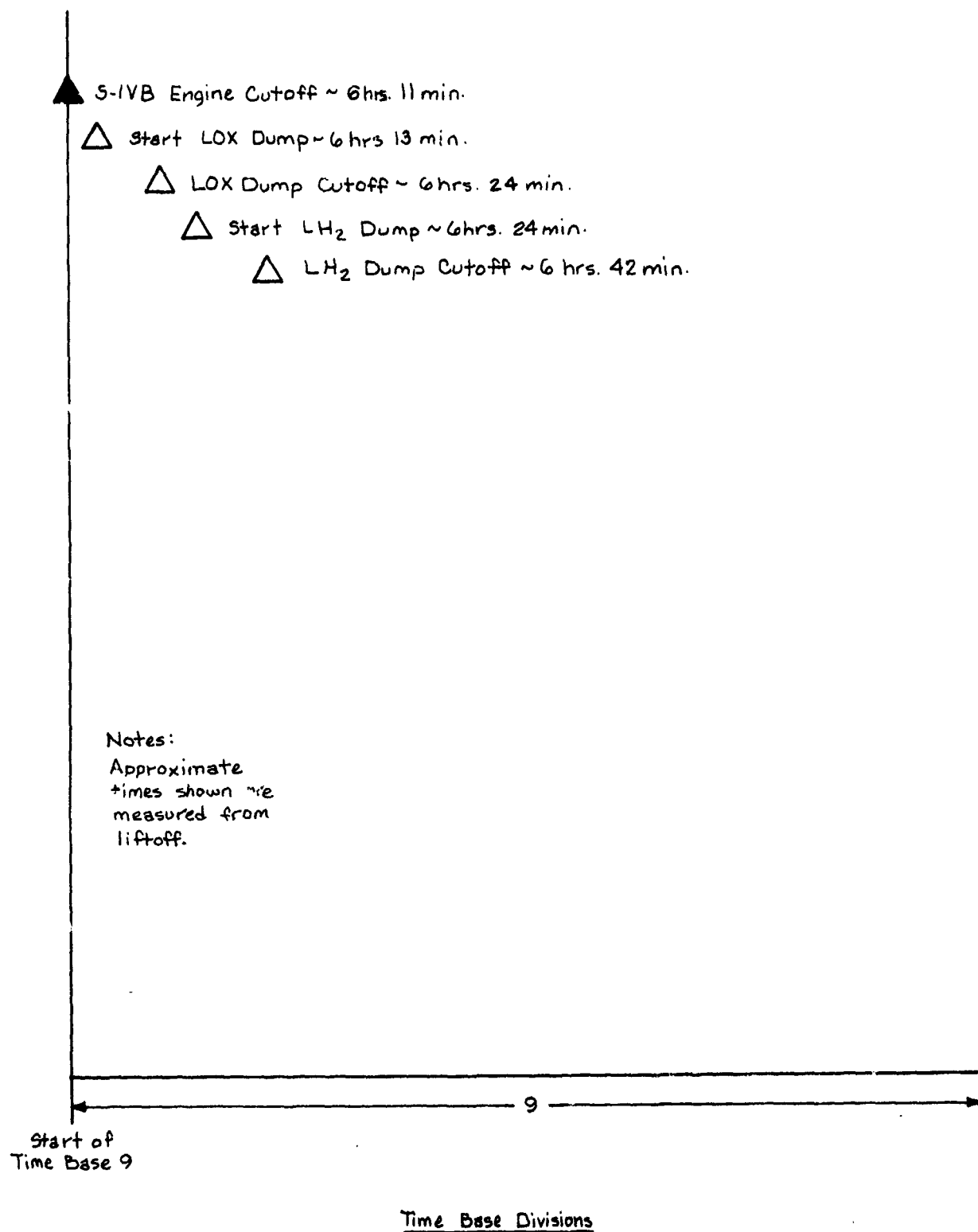
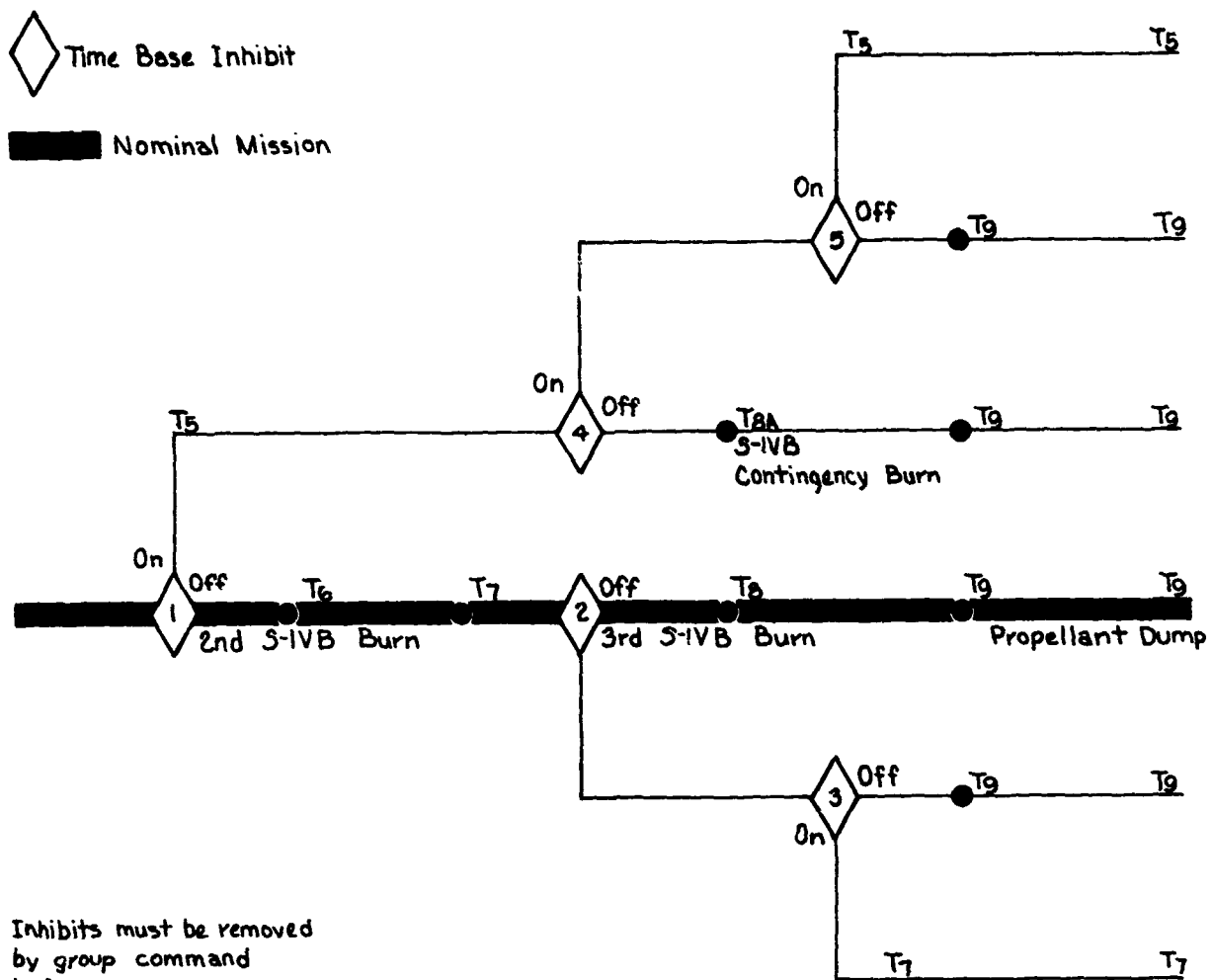


Figure 11

S-IVB Stage  
Flight Sequencing

Reference Event	<u>Time Base References</u>		Comments
	Time Base	G.E.T. Hr/Min/Sec	
Guidance Reference Release	T <sub>0</sub>	- 0:00:17	Initiated by turn 181 count sequencer
Liftoff (I.U. Umbilical Release)	T <sub>1</sub>	0:00:00	Initiated by deactuation of I.U. liftoff relay at umbilical disconnect or vertical acceleration
S-IC Center Engine Cutoff	T <sub>2</sub>	0:02:14	Initiated by S-IC Inboard engine cutoff command from LVDC
S-IC Outboard Engine Cutoff	T <sub>3</sub>	0:02:40	Initiated by the propellant depletion sensors or the thrust-OK switches
S-II Engines Cutoff	T <sub>4</sub>	0:08:51	Initiated by the propellant depletion sensors or the thrust-OK switches
First S-IVB Engine Cutoff	T <sub>5</sub>	0:10:48	Initiated by any two of four functions; S-IVB velocity cutoff issued by the LVDC thrust-OK switches (2) or accelerometer reading
Initiation-Restart Sequence	T <sub>6</sub>	4:36:12	Initiated when LVDC solves the restart equation; inhibit must be removed by ground command
Second S-IVB Engine Cutoff	T <sub>7</sub>	4:46:52	Initiated by any two of four functions; S-IVB cutoff issued by the LVDC, thrust-OK switches (2), or accelerometer reading
Initiation-Second Restart Sequence	T <sub>8</sub>	5:59:35	Initiated a fixed time from initiation of T <sub>6</sub> ; T <sub>7</sub> must also have been accomplished
Third S-IVB Engine Cutoff	T <sub>9</sub>	6:11:14	Initiated by any two of four functions; S-IVB cutoff issued by LVDC, thrust-OK switches (2), or accelerometer
Early S-II/S-IVB Staging	T <sub>4a</sub>	Variable	Initiated by S-II/S-IVB staging switch in the spacecraft during S-II burn
S-IVB Burner Malfunction	T <sub>6a</sub>	Variable	Initiated by burner malfunction signal from S-IVB stage--T <sub>6</sub> +48 seconds to T <sub>6</sub> +496.9 seconds
Delayed Initiation Restart Sequence	T <sub>8a</sub>	6:10:16	Initiated when T <sub>6</sub> is not accomplished. Inhibit must be removed by ground command



Inhibits must be removed by group command before:

- ◇ 1 4:36:02 G.E.T.
- ◇ 2 5:59:25 G.E.T.
- ◇ 3 6:24:55 G.E.T.
- ◇ 4 6:10:06 G.E.T.
- ◇ 5 6:18:39 G.E.T.

Figure 12

Time Base Inhibit



## GUIDANCE AND CONTROL SYSTEM (G&C)

### Function and Description

The G&C system provides the following basic functions during flight:

1. Stable positioning of the vehicle to the commanded position with a minimum amount of sloshing and bending.
2. A first stage tilt attitude program which gives a near zero lift trajectory through the atmosphere.
3. Provides steering commands during S-II and S-IVB burns which guide the vehicle to a predetermined set of end conditions while maintaining a minimum propellant trajectory for earth orbit insertion.
4. Maintains the proper vehicle position during earth orbit.
5. Provides guidance during the second S-IVB burn, placing the vehicle in the proper waiting orbit.

### G&C Hardware

The Stabilized Platform (ST-124M) is a three gimbal configuration with gas bearing gyros and accelerometers mounted on the stable element. Gimbal angles are measured by redundant resolvers and inertial velocity is obtained from integrating accelerometers (see figure 13).

The Launch Vehicle Data Adapter (LVDA) is an input-output device for the Launch Vehicle Digital Computer (LVDC). The LVDA/LVDC components are digital devices which operate in conjunction to carry out the flight program. The flight program performs the following functions: (1) processes the inputs from the ST-124M, (2) performs navigation calculations, (3) provides the first stage tilt program, (4) calculates IGM steering commands, (5) calculates attitude errors, (6) issues launch vehicle sequencing signals.

The Control/EDS Rate Gyro Package contains nine rate gyros (triple redundant in three axes). Their outputs go to the Control Signal Processor (CSP) where they are voted and sent to the Flight Control Computer (FCC) for damping vehicle angular motion.

The FCC is an analog device which receives attitude error signals from the LVDA/LVDC and vehicle angular rate signals from the CSP. These signals are filtered and scaled, then sent as commands to the S-IC, S-II, and S-IVB engine actuators and to the Auxiliary Propulsion System (APS) Control Relay Packages. The Control Relay Packages accept FCC commands and relay these commands to operate propellant valves in the APS.

The Switch Selectors in each stage are used to relay Sequencing Commands from the LVDA/LVDC to other locations in the vehicle.

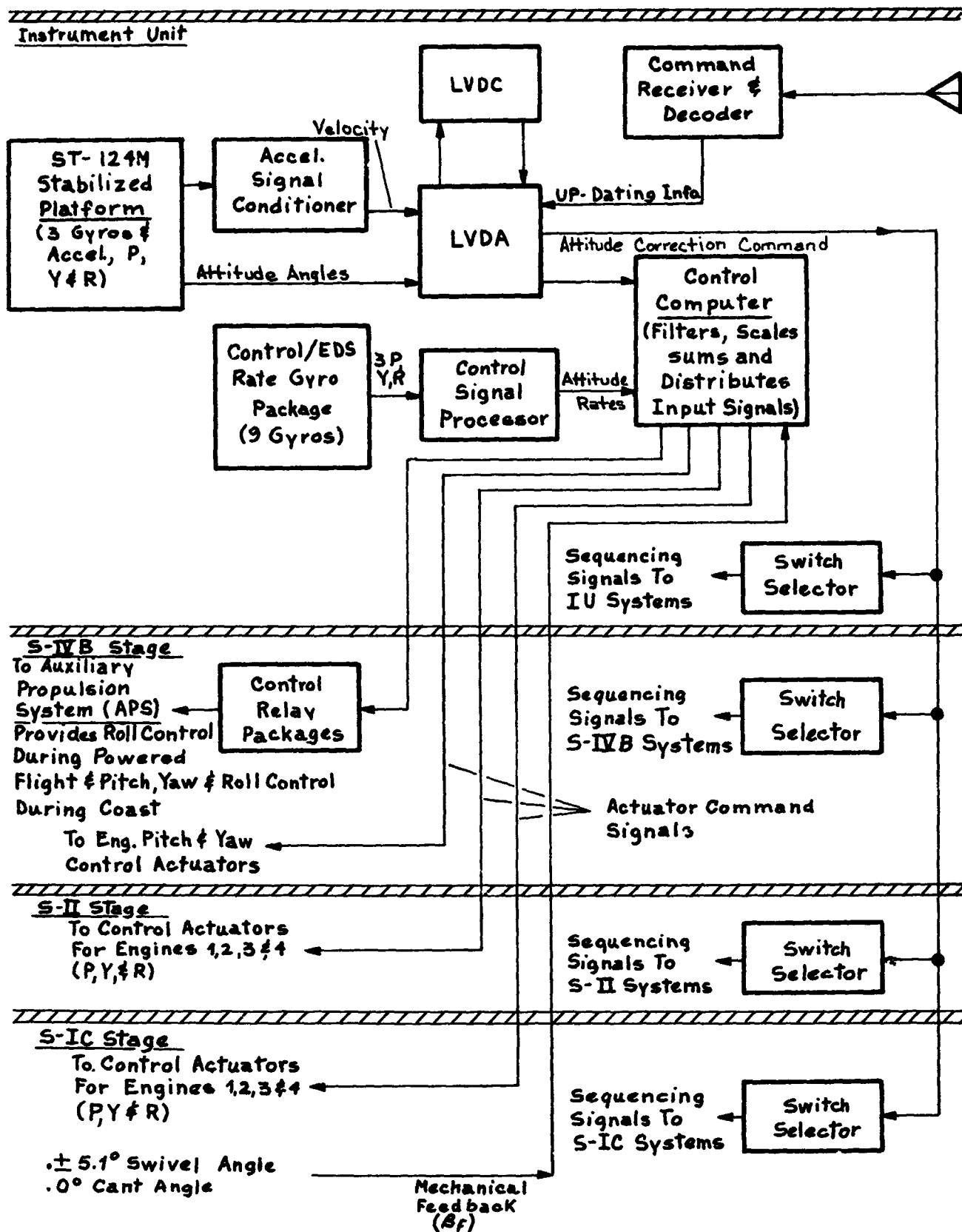


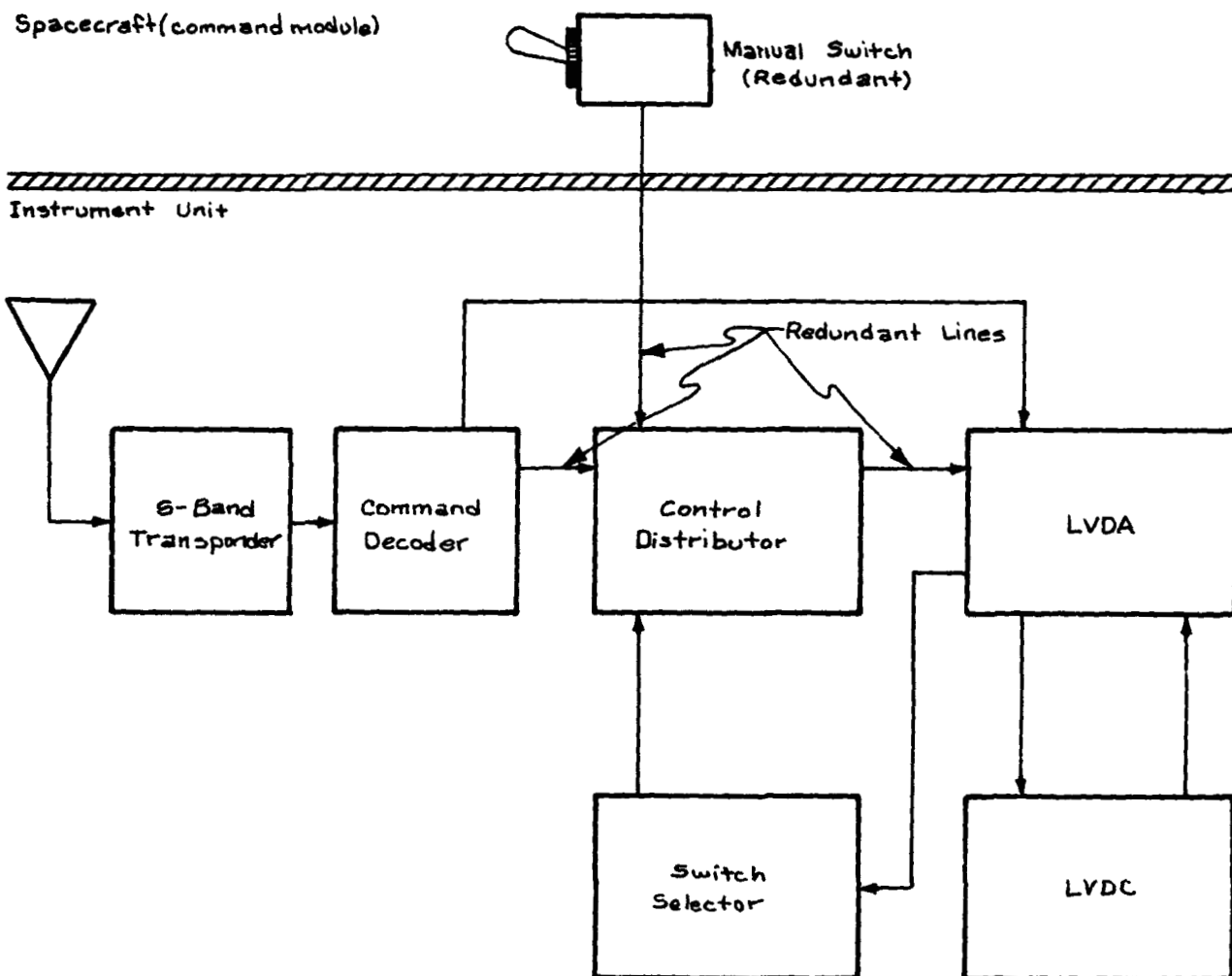
Figure 13

## DIGITAL COMMAND SYSTEM CAPABILITY:

The following summary describes the AS-503 Digital Command Systems' capability:

<u>Function</u>	<u>Description</u>	<u>Periods of Acceptance</u>
Inhibit maneuver	Coast phase attitude maneuver inhibit	From T5 + 0 seconds until T6 - 9 or T5 to T8 - 9 or T5 to T9 - 1
Maneuver update	Time change to start coast phase maneuver	From T5 + 100 to T6 - 9 and T7 + 15 to T8 - 9 and from T9 + 15 to EOM
Time base update	Time base time is advanced or retarded	From T5 to T6 - 9 and T7 to T8 - 9 and from T9 to EOM
Generalized switch selector	Specified switch selector function is issued at the first opportunity	From T5 + 0 seconds to T6 + 570 and from T7 to T8 + 450 and from T9 to EOM
Sector dump	Contents of specified memory location are telemetered	From T5 + 100 seconds to T6 - 9 and from T7 + 15 seconds to T8 - 9 and from T9 + 15 to EOM
Telemeter single memory location	Contents of specified memory location are telemetered	From T5 + 100 seconds to T6 - 9 and from T7 + 15 seconds to T8 - 9 and from T9 + 15 to EOM
Terminate	Stop DCS processing and reset for a new command	Anytime any other DCS command is acceptable
Inhibit water control valve logic	Inhibit water valve from changing position	From T5 + 0 seconds to T6 - 9 and from T7 + 0 and T8 - 9 and from T9 to EOM
Switch antenna to omni, low gain, or high gain	Both PCM and CCS antennas are switched with these commands	From T5 + 100 seconds to T6 - 9 and from T7 + 15 to T8 - 9 and from T9 + 15 to EOM
Restart maneuver enable	Remove programmed inhibit of maneuvers	From T5 to T6 - 9 and from T7 to T8 - 9; or from T5 to T8A - 9; or from T5 (or T7) to T9 - 1

EOM -- End of mission



DCS hardware can be enabled by spacecraft manual switch for command action during coast phase operation or prior to separation. However, commands will only be accepted by the flight program within the period of time programmed in the LVDC, as described on page 26.

Figure 14

Digital Command  
System

## INSTRUMENTATION SYSTEMS

The Saturn V Instrumentation Systems are functionally divided into three parts on each stage. These separate divisions or subsystems are:

- Measuring Systems
- Telemetry Systems
- RF and Tracking Systems

### Measuring

The purpose of the measuring systems is to detect the phenomena to be measured and to process and distribute this data to the input of each stage telemetry system. All measurements, regardless of their original characteristics, must be processed into electrical signals within a 0 to 5-volt range prior to delivery to the stage telemetry system. The telemetry system accepts these input signals for transmission to the ground recovery stations.

The following table contains a measurement breakdown for the launch vehicle and the spacecraft.

### Telemetry

The Telemetry System for each stage of the vehicle must accept signals produced by the measuring portion of the instrumentation system, and accurately reproduce and transmit them to the ground stations. Measurement signals are accepted at a fixed input level, processed, and fed to the proper airborne antennas. In the case of checkout measurements, the signals are transmitted via breakaway cable arrangement to the ground checkout station prior to liftoff.

### RF and Tracking

The Vehicle RF and Tracking Systems are described and illustrated on pages 30 and 31.

<u>Measurement Summary-L/V</u>					
Measurement Designation	S-1C Stage	S-11 Stage	S-1VB Stage	Inst. Unit	L/V Total
Acceleration	3	11	-	4	18
Acoustic	5	5	-	-	10
Temperature	190	330	75	16	611
Pressure	177	188	79	10	454
Vibration	70	63	-	-	133
Flow Rate	35	10	4	4	53
Position	1	44	8	21	74
Signals	132	223	72	105	532
Liquid Level	22	6	7	-	35
Voltage, Current, Frequency	11	65	38	38	152
Miscellaneous	12	4	9	-	25
Angular Velocity	3	3	-	24	30
Strain	-	27	-	-	27
RPM	5	10	2	-	17
Guidance and Control	-	-	-	60	60
RF and Telemetry	-	-	-	14	14
Totals	666	989	294	296	2245
ESE Display	97	82	100	177	456
Auxiliary Display	64	81	63	18	226
Flight Control	28	80	86	104	298

<u>Measurement Summary-S/C</u>				
	<u>CM</u>	<u>SM</u>	<u>LM</u>	<u>S/C Total</u>
Pressure	18	37	103	158
Temperature	19	43	153	215
Discrete Event	84	5	303	392
Voltage, Current, Frequency	44	3	206	253
Miscellaneous	30	55	115	200
Totals	195	143	880	1218

Figure 15

AS-504  
Measurement Summary

## VEHICLE TRACKING SYSTEMS

In the Saturn V Space Vehicle there is a continuous requirement to transmit information to ground stations in order to track the vehicle. This requirement is filled by the RF Systems.

The RF System functions to transmit (via RF carrier) all vehicle flight evaluation data as well to evaluate vehicle performance (flight path) for ground receiving stations. These functions are accomplished through the use of Antenna and Tracking Systems.

The principal tracking systems used are:

- ODOP (offset doppler) system - Used in the S-IC stage
- C-band radar - Used in the IU and spacecraft
- Unified S-band system - Used in the spacecraft.

### ODOP System (S-IC)

An offset doppler, frequency measurement system is an elliptical tracking system which measures the total doppler phase shift in a ultra-high frequency (UHF) continuous wave (CW) signal transmitted to the S-IC stage. The system uses a fixed station (ground) transmitter, a vehicle-borne transponder and three or more fixed station (ground) receivers.

### C-Band (IU and SC)

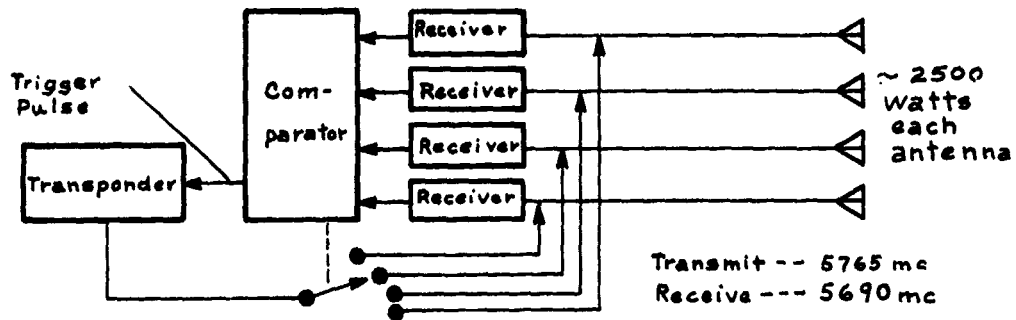
C-Band is a pulse radar system which is used for precise tracking during launch and orbit phases. Two C-Band radar transponders carried in IU to provide radar tracking capabilities independent of vehicle attitude.

### Unified S-Band System (SC)

The Unified Side Band (USB) System provides tracking capability to the USB ground stations.

## Spacecraft

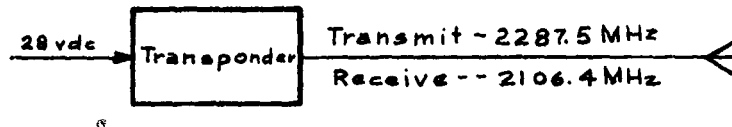
### C-Band Radar System



- Crystal switch driven by Comparator
- Automatically selects strongest receivers' antenna for output

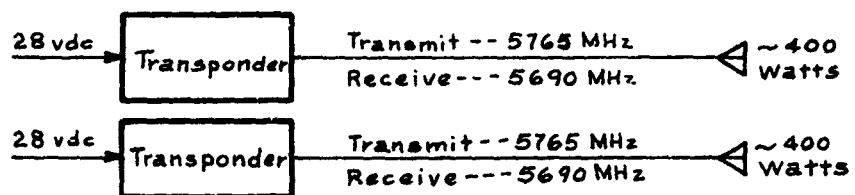
Note:  
Different Pulse Code  
than C-Band in IU.

### Unified S-Band System



## Instrument Unit

### C-Band Radar System



## S-IC Stage

### ODOP System (Offset Doppler Velocity and Position)

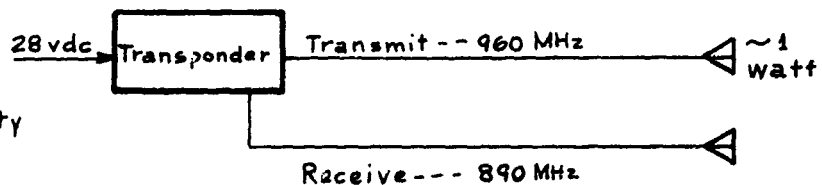


Figure 16

Vehicle Tracking Systems



## SPACE VEHICLE WEIGHT VS. FLIGHT TIME

Mainstage propellant consumption during S-IC stage powered flight (approximately 159 seconds) is 4,660,200 pounds. Propellant consumption during S-II stage powered flight (approximately 372 seconds) is approximately 972,200 pounds and during S-IVB stage powered flight, including first ignition and subsequent second and third ignitions, (approximately 422 seconds) is approximately 230,700 pounds.

### VEHICLE WEIGHT DATA (Approximate)

### Pounds

Total at S-IC stage ignition	6,483,200
Total at holddown arm release	6,396,900
Total at S-IC stage O.E.C.O.	1,827,900
Total at S-II stage ignition	1,449,400
Total at S-II stage E.C.O.	463,400
Total at S-IVB stage first ignition	358,200
Total at S-IVB stage E.C.O.	289,900
Total at spacecraft separation	195,000
Total at S-IVB stage second ignition	192,200
Total at S-IVB stage second E.C.O.	163,600
Total at S-IVB stage thrid ignition	162,500
Total at S-IVB stage third E.C.O.	34,400

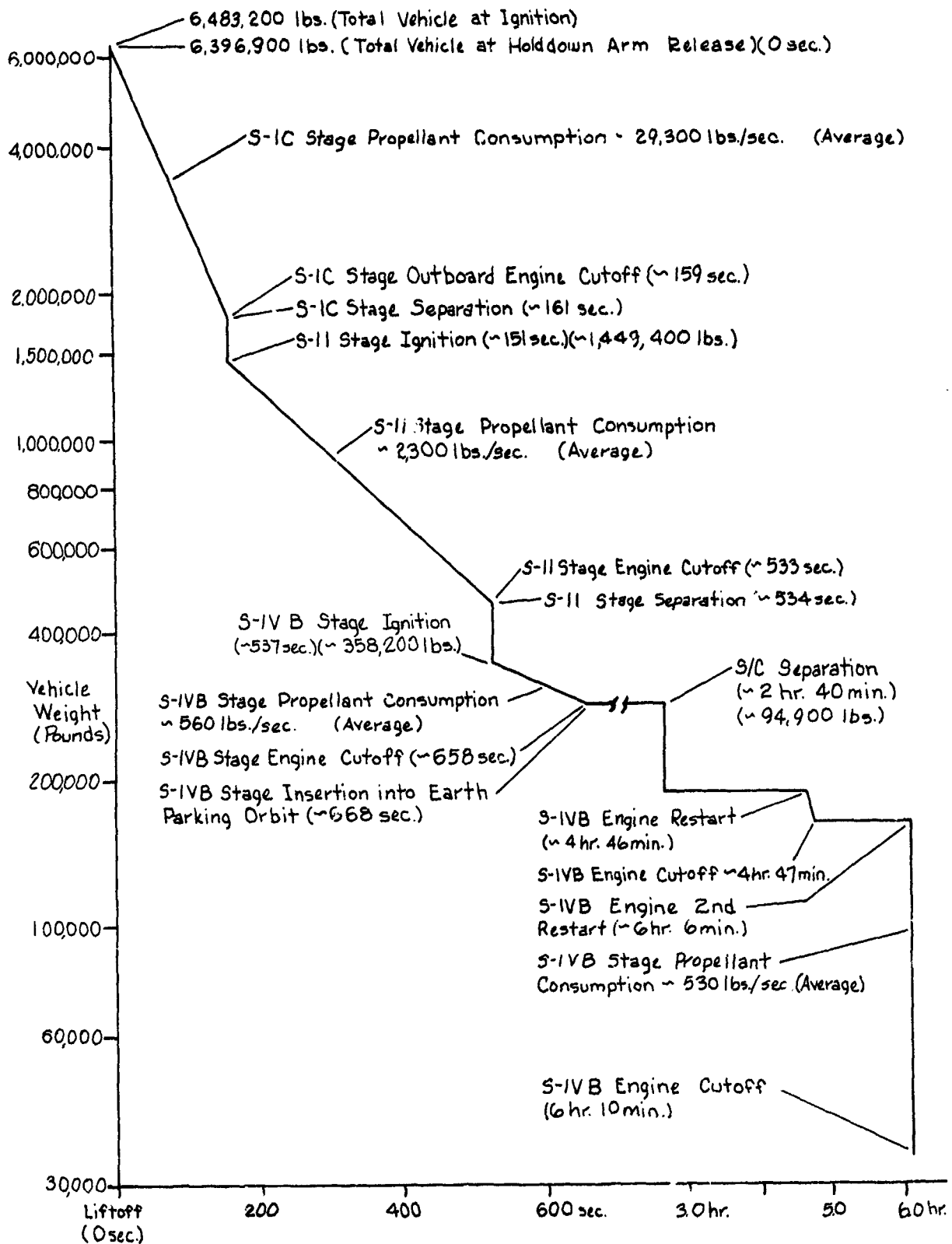


Figure 17

Space Vehicle Weight  
Vs. Flight Time

### **S-IC STAGE STRUCTURE**

The S-IC stage is approximately 138 feet long and 33 feet in diameter and is powered by five liquid-fueled Rocketdyne F-1 engines which generate a nominal thrust of 7,610,000 pounds. Stage engines are supplied by a bi-propellant system of liquid oxygen (LOX) as the oxidizer and RP-1 as the fuel.

The stage interfaces structurally and electrically with the S-II stage (forward skirt structure).

Mounted on the structural airframe, the stage consists of an RP-1 fuel tank, a LOX tank, five F-1 engines, electrical and pneumatic control as well as emergency flight termination equipment. Eight retro rockets, used during S-IC/S-II stage separation, will cause the S-IC stage to back away from the flight vehicle when fired.

TV Camera and Film Camera Systems are not installed in the S-IC stage for AS-504.

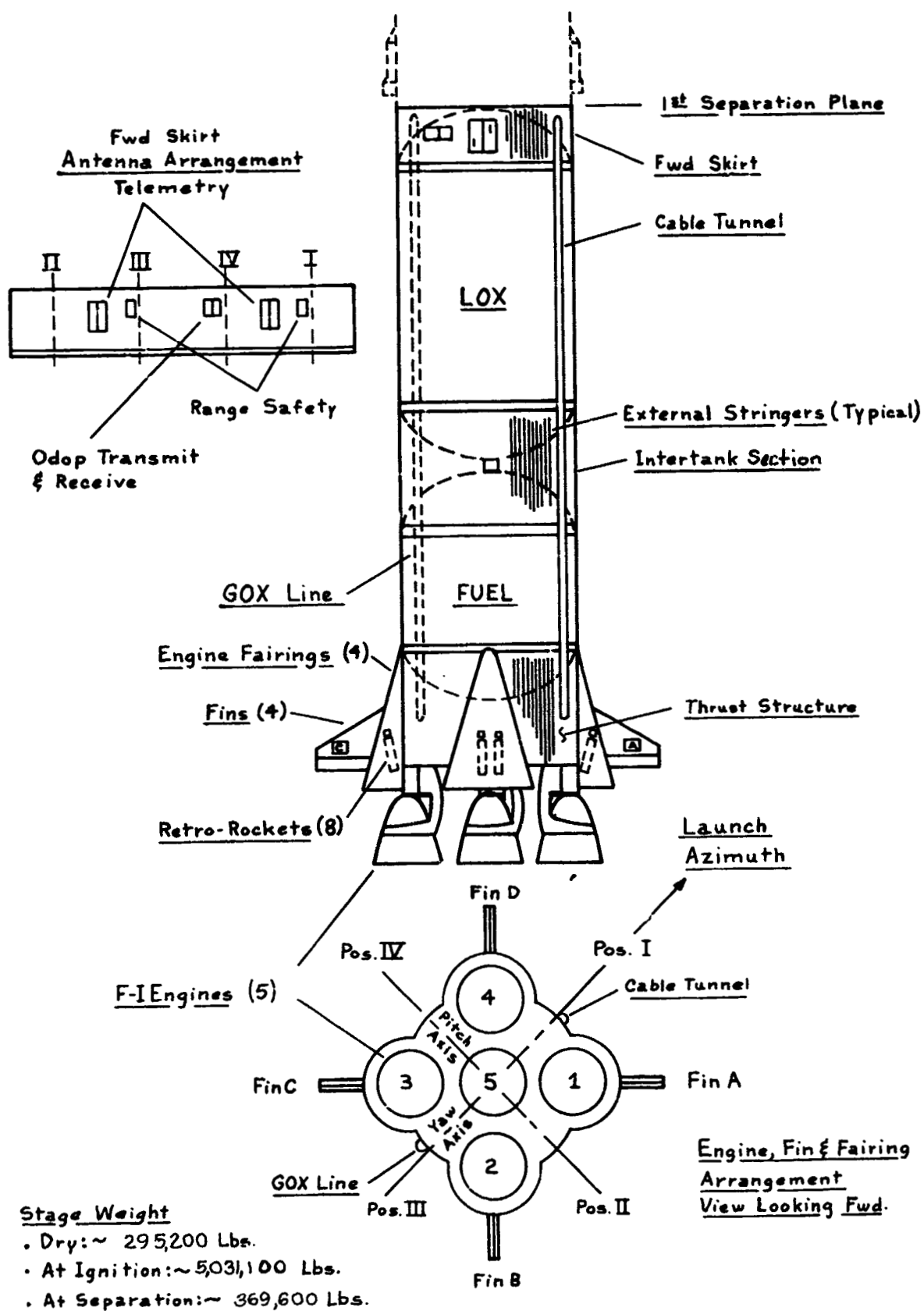


Figure 18

S-IC Stage Configuration

## F-1 ENGINE OPERATION

The F-1 engine is started by ground support equipment. Ground fluid pressure opens ports in the main LOX valves. Opening of the main LOX valves admits LOX under tank pressure to the thrust chamber and allows control fluid to enter the gas generator. Opening of the gas generator valve permits LOX and RP-1 to enter the gas generator combustion chamber where it is ignited and the hot gases are discharged into the thrust chamber where they are ignited by the turbine exhaust igniters. When the RP-1 reaches approximately 375 psig a valve in the hypergol cartridge opens allowing LOX and RP-1 to build up pressure against the hypergol burst diaphragm. At approximately 500 psig the diaphragm ruptures allowing hypergol and RP-1 to enter the thrust chamber causing spontaneous combustion upon contact with the LOX, thereby establishing primary ignition. As thrust pressure builds up the RP-1 valves open admitting RP-1 to the thrust chamber and the transition to mainstage operation.

The inboard engine is cutoff by a signal from the IU. Outboard engines are cutoff by optical type LOX depletion sensors with fuel depletion sensors as backup. A command from the IU supplies a command to the switch selector to enable the outboard engine cutoff circuitry. When two or more of the four LOX level sensors are energized, a timer is activated. Expiration of the timer energizes a stop solenoid for each engine which energizes the main LOX and main RP-1 valves. The sequence closing of the main LOX valve followed by sequence closing of the main RP-1 valve interrupts propellant flow and terminates engine operation.

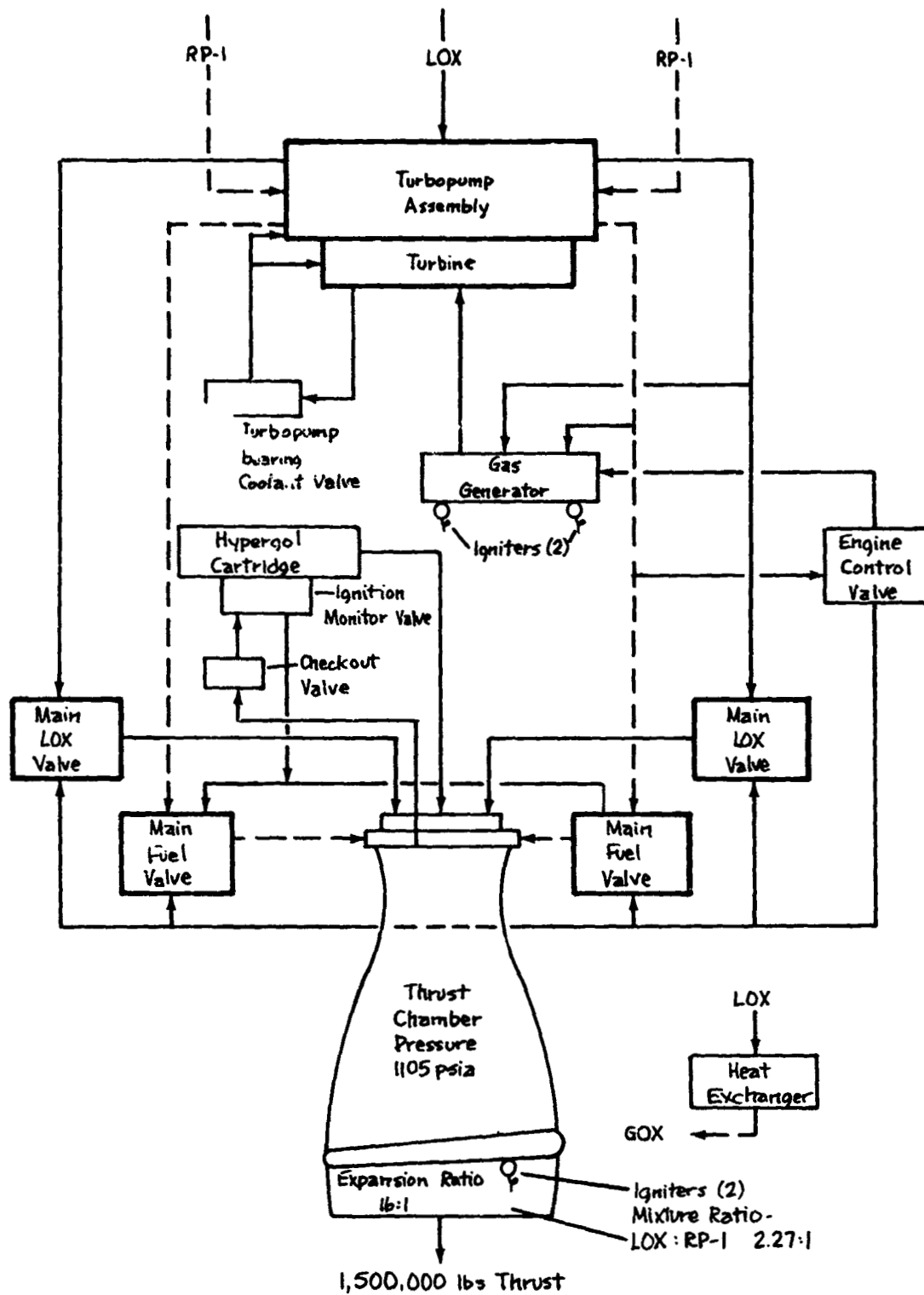


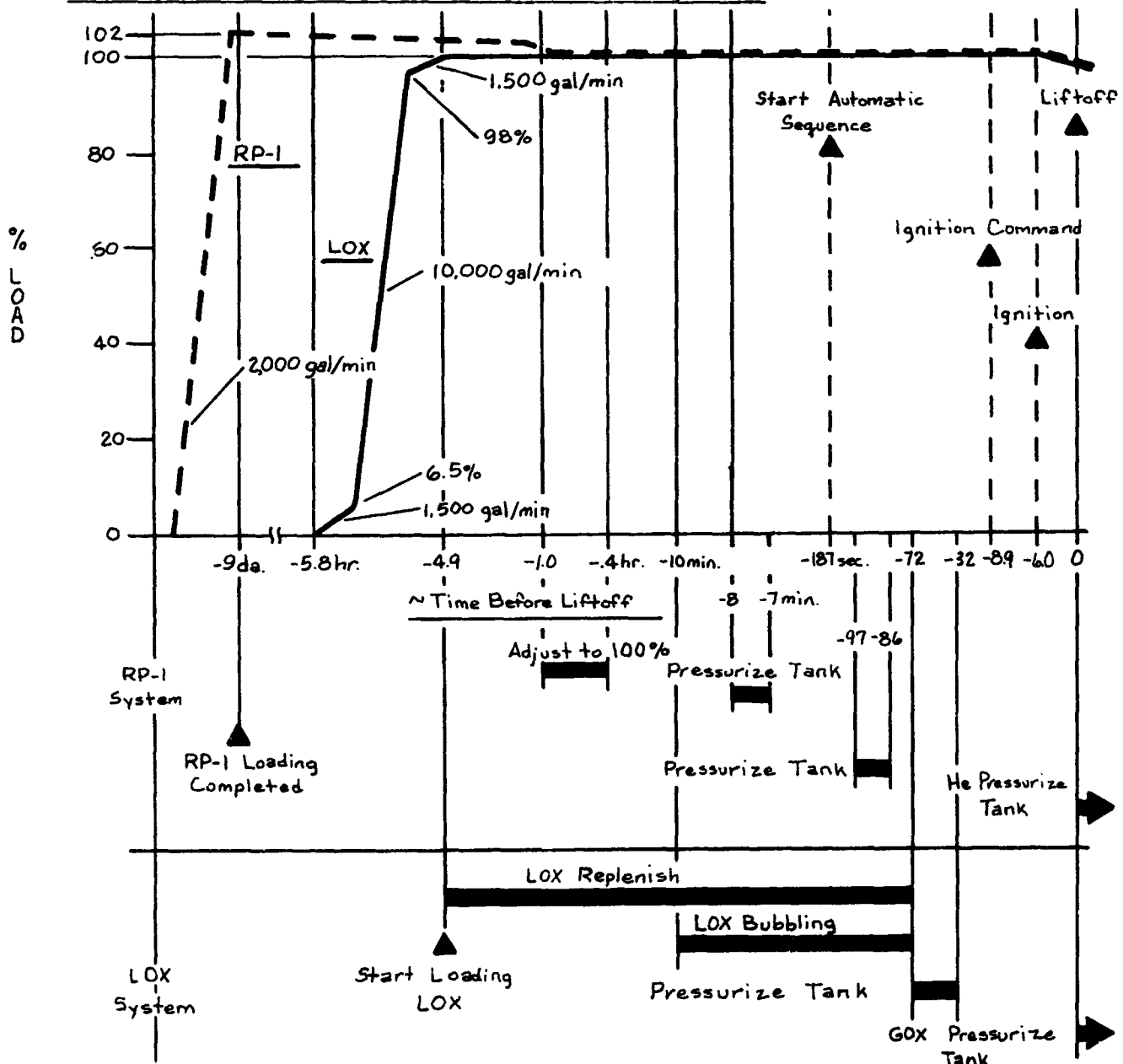
Figure 19

F-1 Engine System

## S-IC STAGE PROPELLANT SYSTEM

The S-IC stage propellant system is composed of one LOX tank, one RP-1 tank, propellant lines, control valves, vents, and pressurization subsystems. Loading of LOX and RP-1 tanks is controlled by ground computers. RP-1 loading is completed approximately nine days prior to liftoff. LOX bubbling, through lines 1 and 3, is started at the beginning of LOX chilldown operation and is continued throughout LOX loading and again before liftoff to prevent possible geysering. Prior to liftoff the RP-1 tank and the LOX tank is pressurized by helium from a ground source. At liftoff the RP-1 tank is pressurized with helium stored in bottles located in the LOX tank and heated by passing the helium through the heat exchanger. LOX tank pressurization is maintained by LOX bled from the engine and converted to GOX in the heat exchanger.

### S-IC PROPELLANT LOAD AND OPERATIONAL SEQUENCE



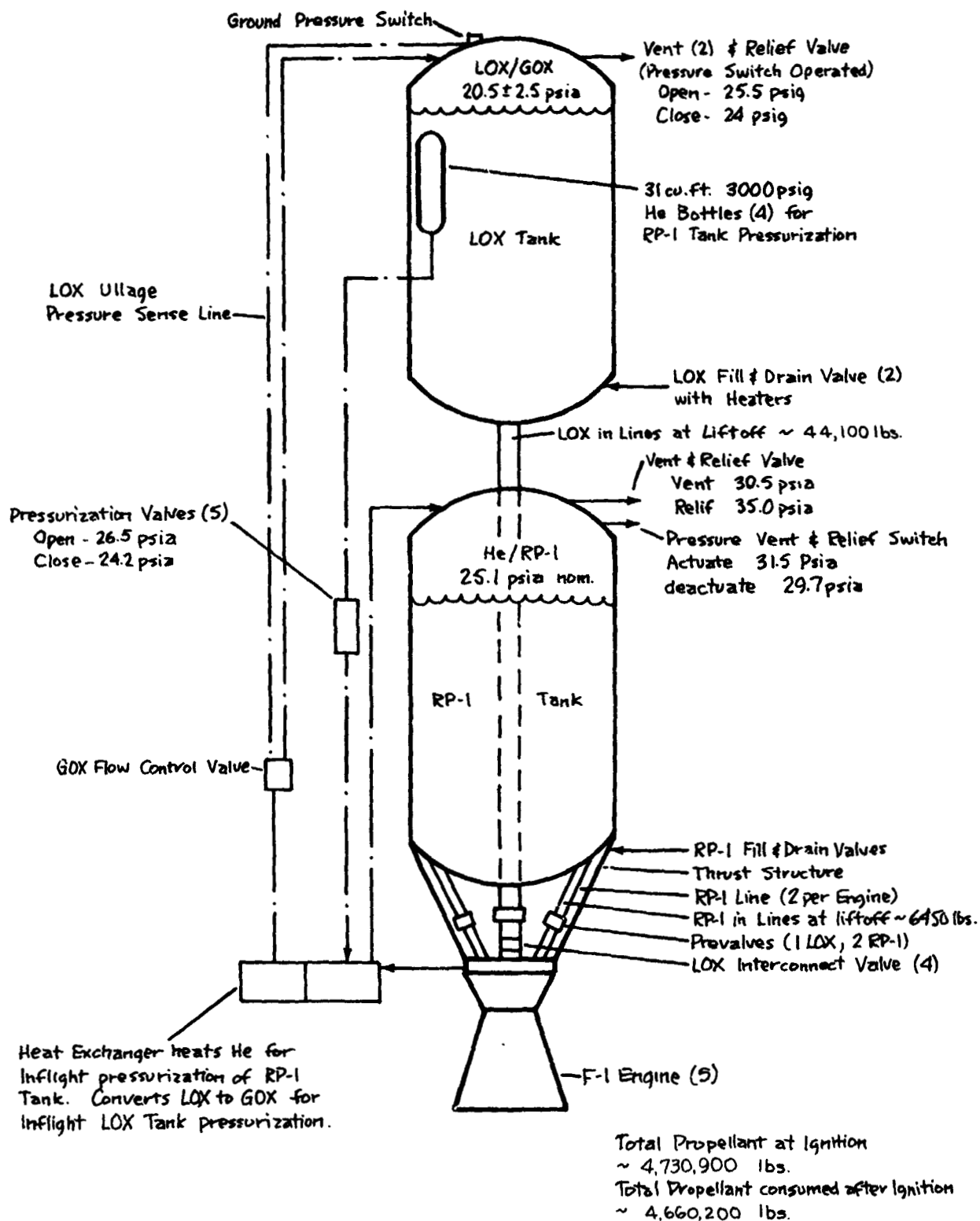


Figure 20

S-IC Stage Propellant System



### **S-IC STAGE THRUST VECTOR CONTROL SYSTEM**

Each of the four outboard F-1 engines is gimbal mounted on the stage thrust structure to provide engine thrust vectoring for vehicle attitude control and steering. Two hydraulic actuators are utilized to gimbal each engine in response to signals from the Flight Control Computer located in the Instrument Unit.

The thrust vector control system is part of the engine system. During engine operation, high pressure control fluid is supplied from the turbopump assembly to the servo valve and actuators. The fluid returns to the inlet of the turbopump assembly.

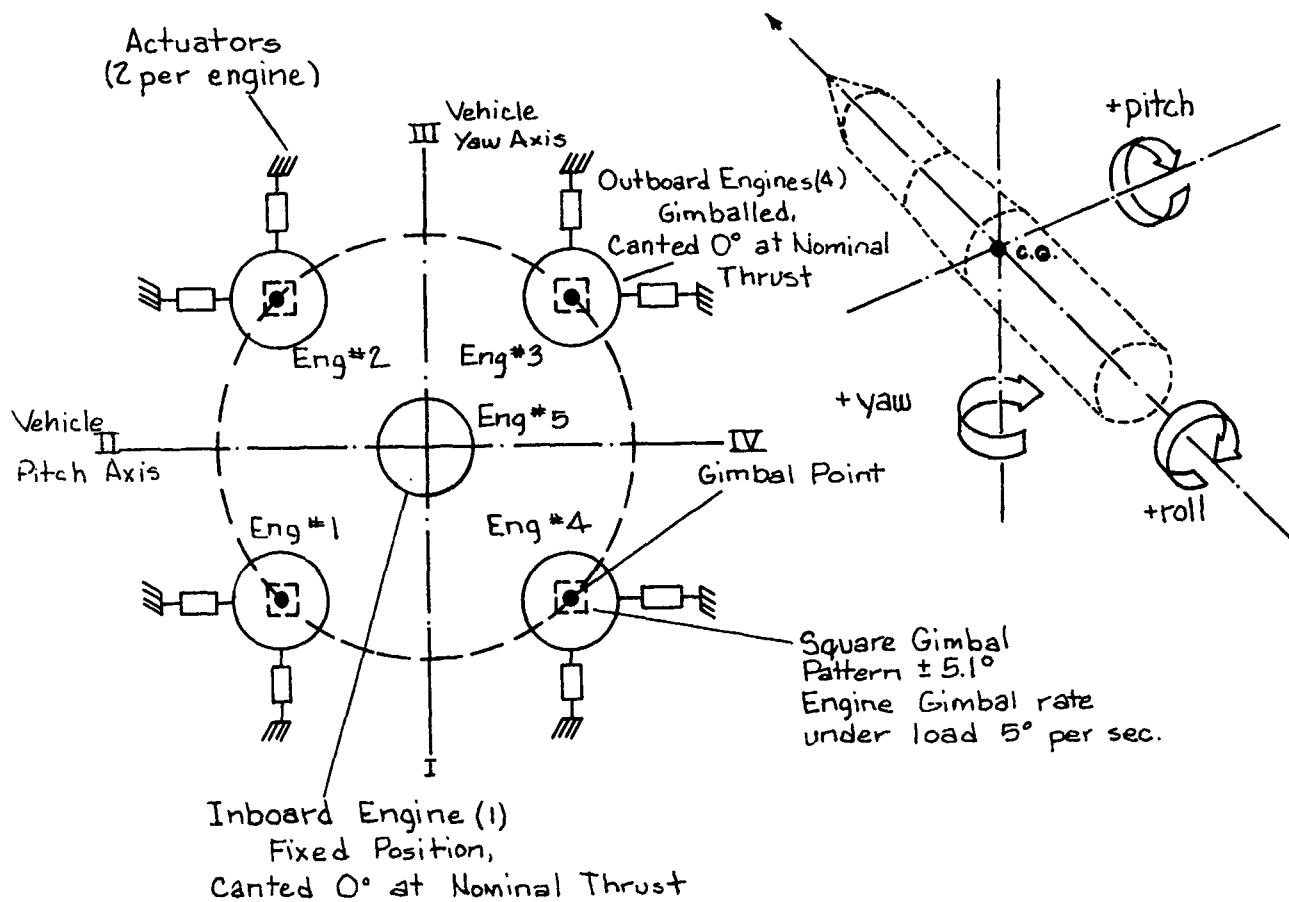
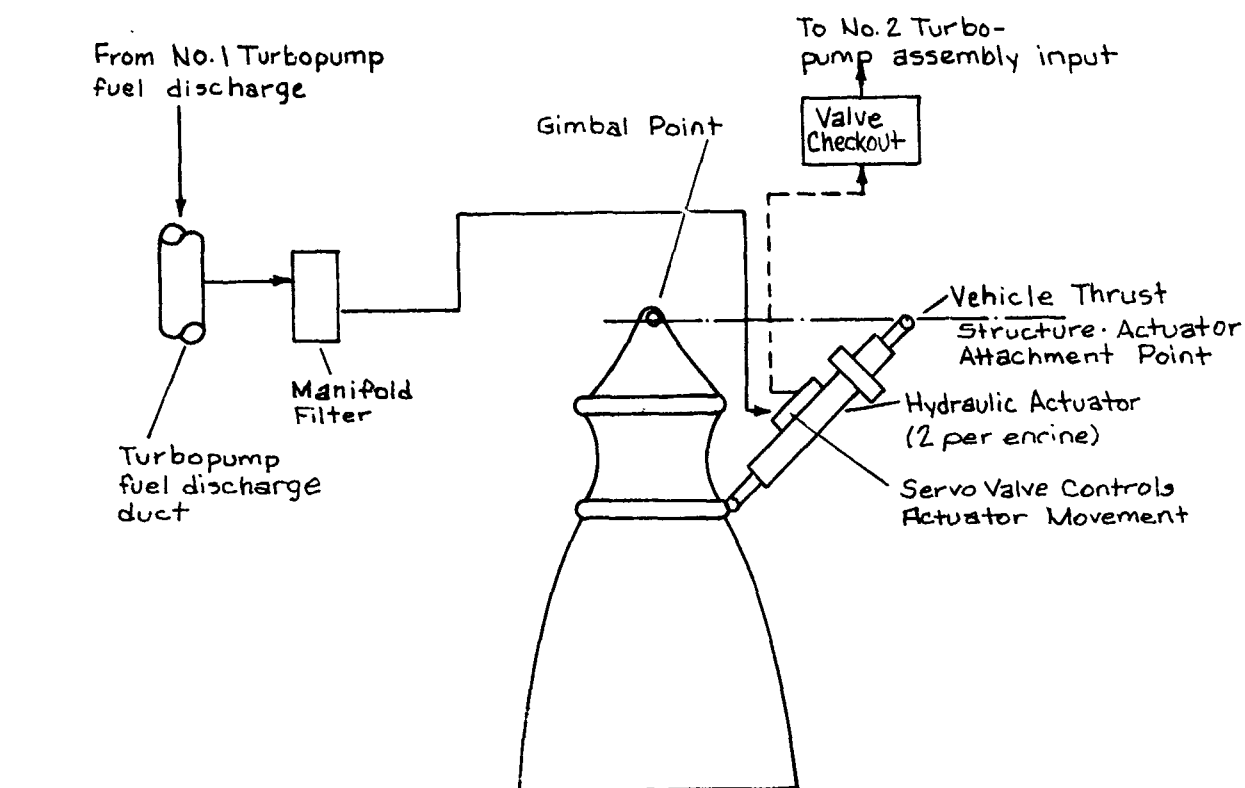


Figure 21

S-IC Stage Thrust Vector Control System

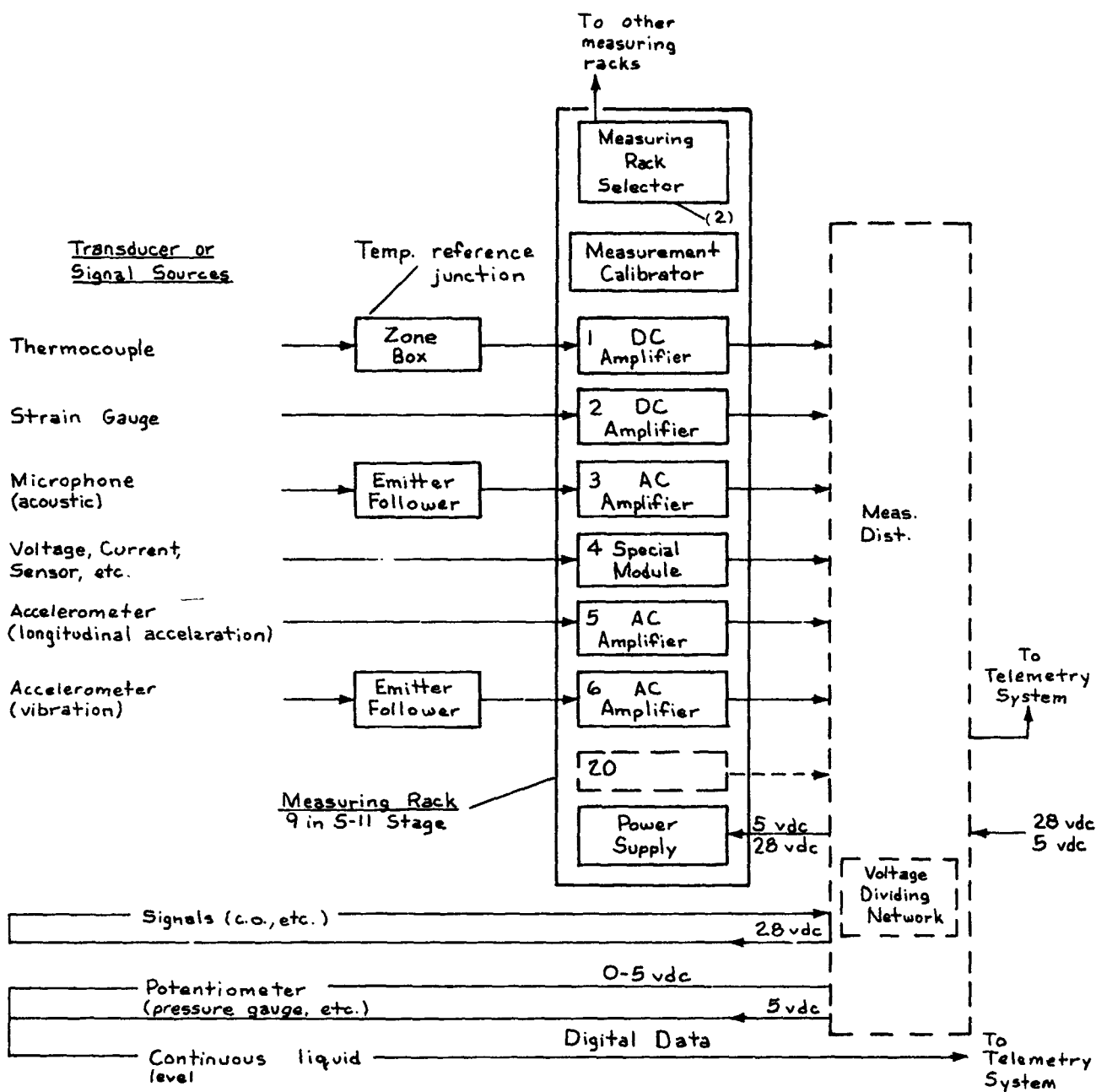


Figure 22

S-1C Stage  
Measuring System

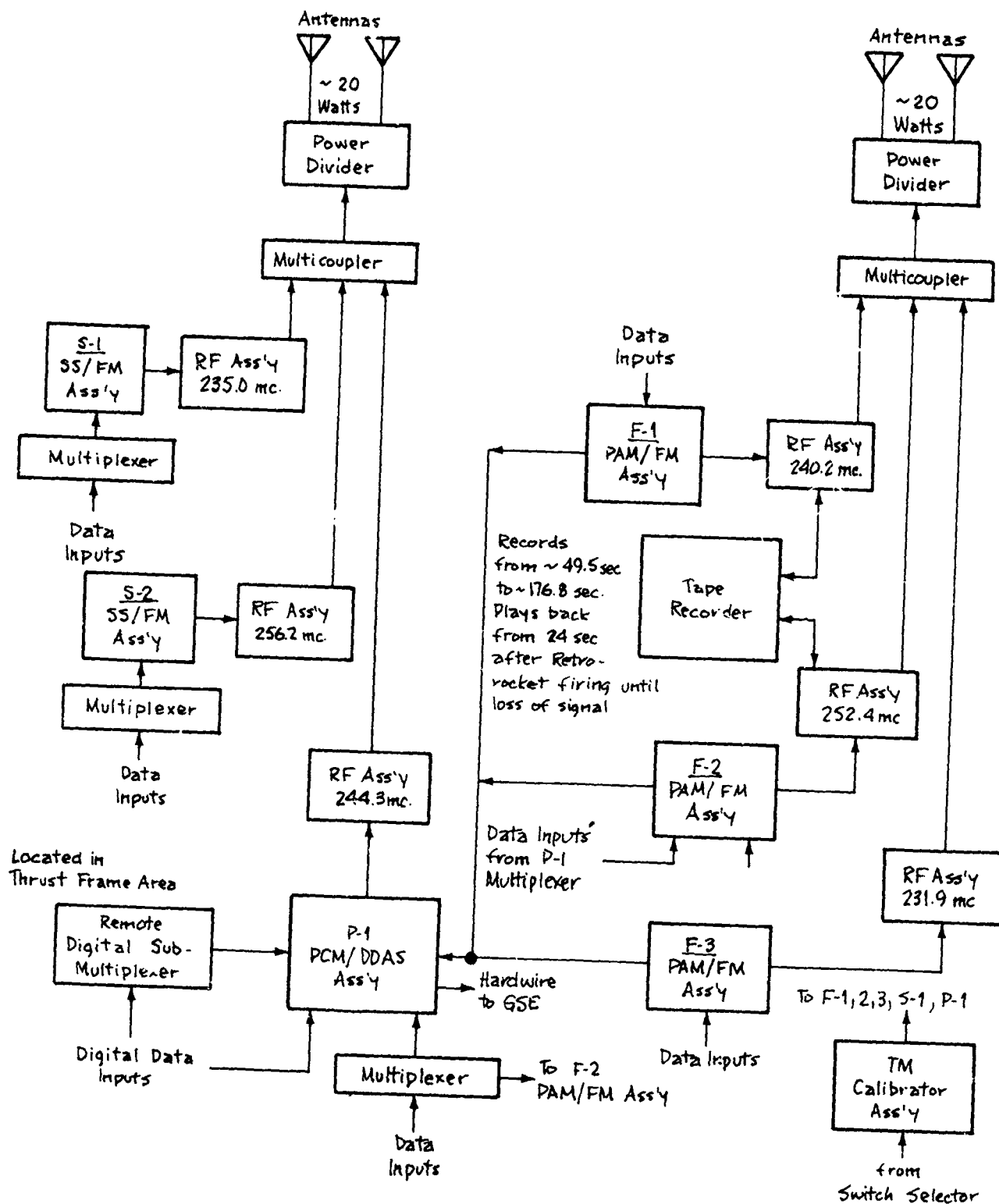


Figure 23

S-IC Stage Telemetry System

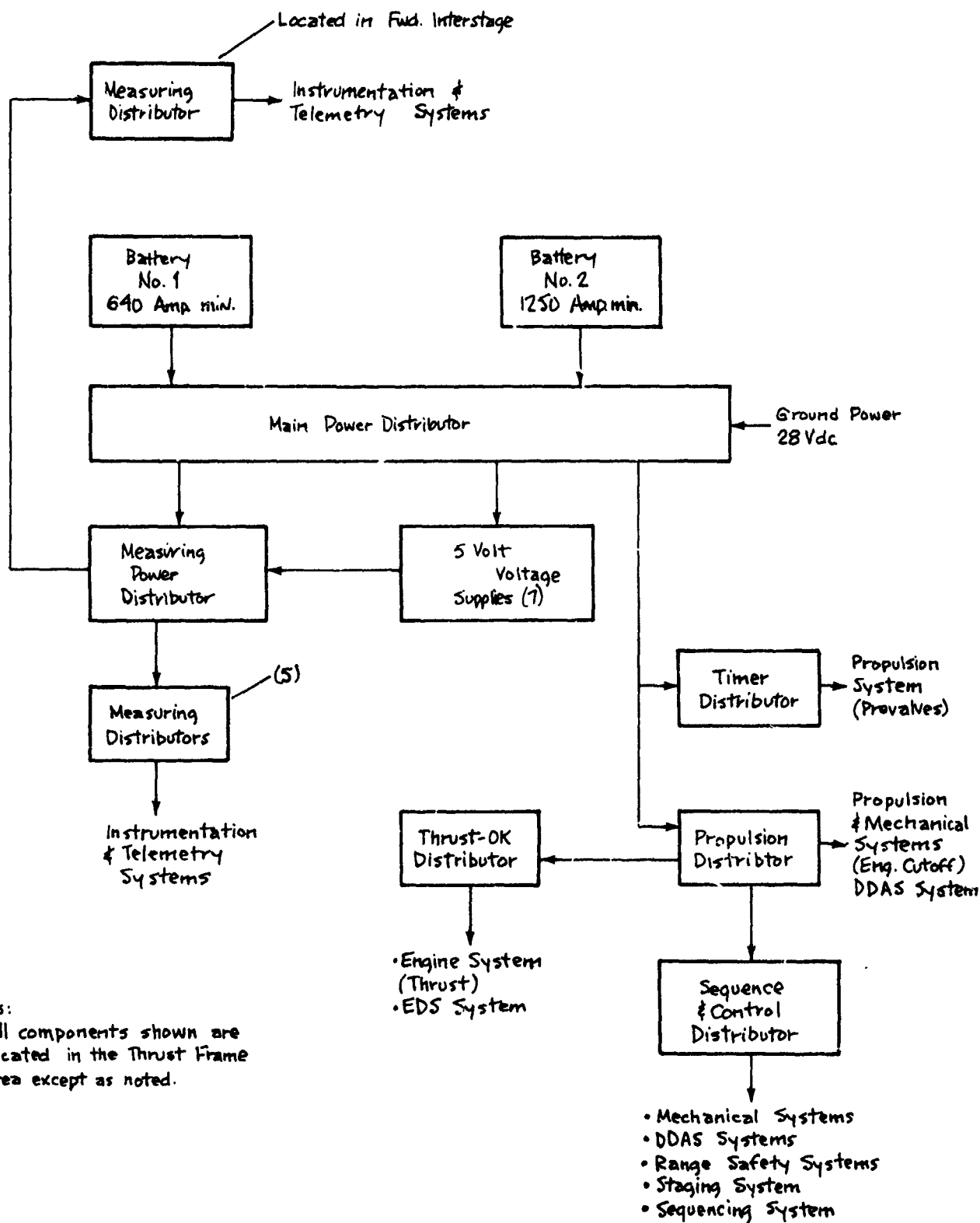


Figure 24

S-IC Stage Electrical Power and Distribution System

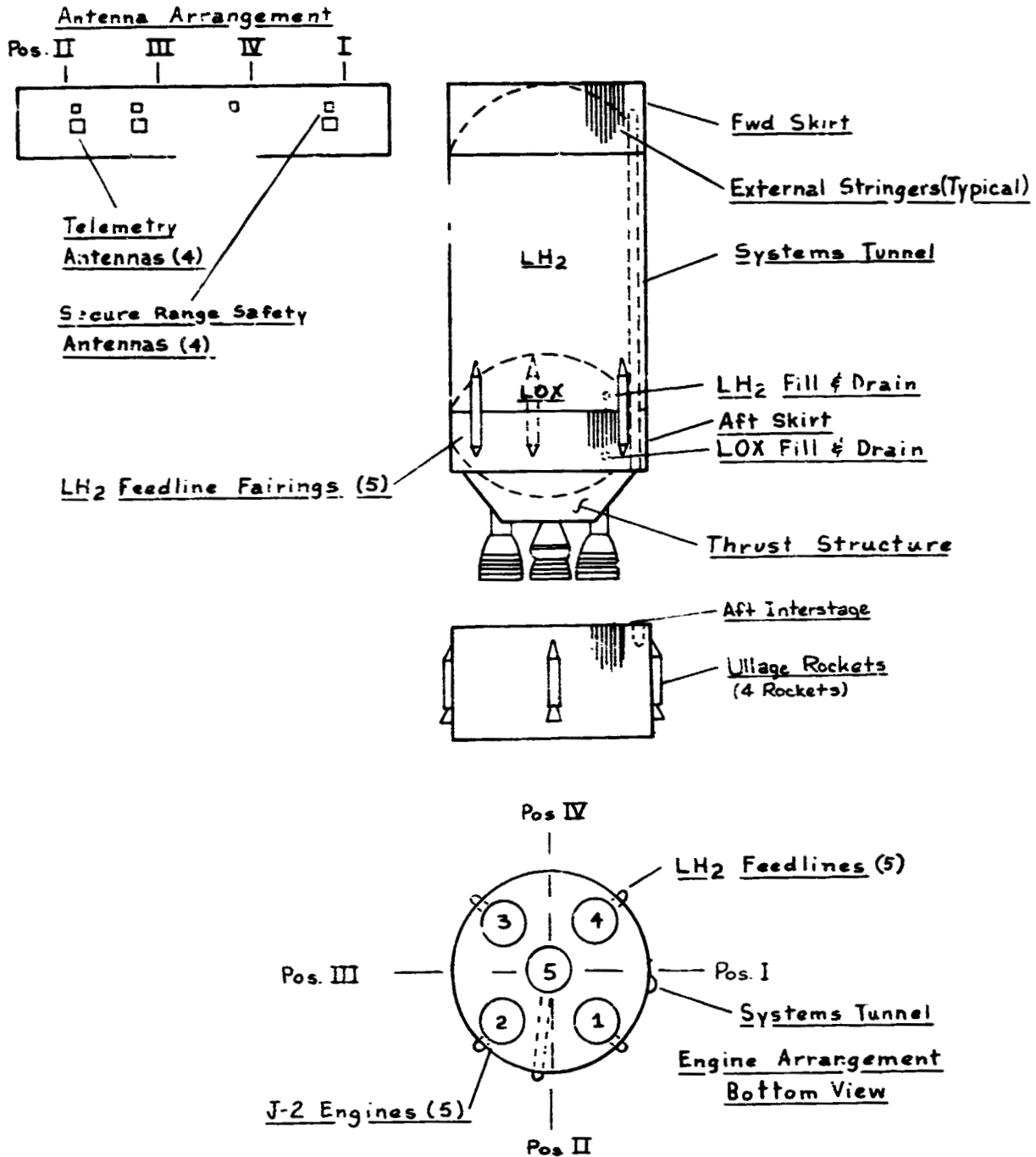
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### S-II STAGE STRUCTURE

The S-II stage is a large cylindrical booster approximately 81 feet in length and 33 feet in diameter. The stage is powered by five liquid propellant J-2 rocket engines which combine to develop a total thrust of 1,140,000 pounds.

In addition to the J-2 rocket engines, the structural air frame of the S-II stage mounts a forward and aft skirt, an aft interstage, a liquid oxygen and liquid hydrogen tank plus the associated piping, valves, wiring, electrical and electronic equipment.

Note: The retro-rockets for S-II Stage separation are located in the S-IVB aft interstage.



#### Stage Weight

- Dry: ~ 84,600 lbs.
- At S-II Ignition: ~ 1,059,600 lbs.
- At S-II Cutoff: ~ 97,000 lbs.
- At S-II Separation: ~ 96,500 lbs.

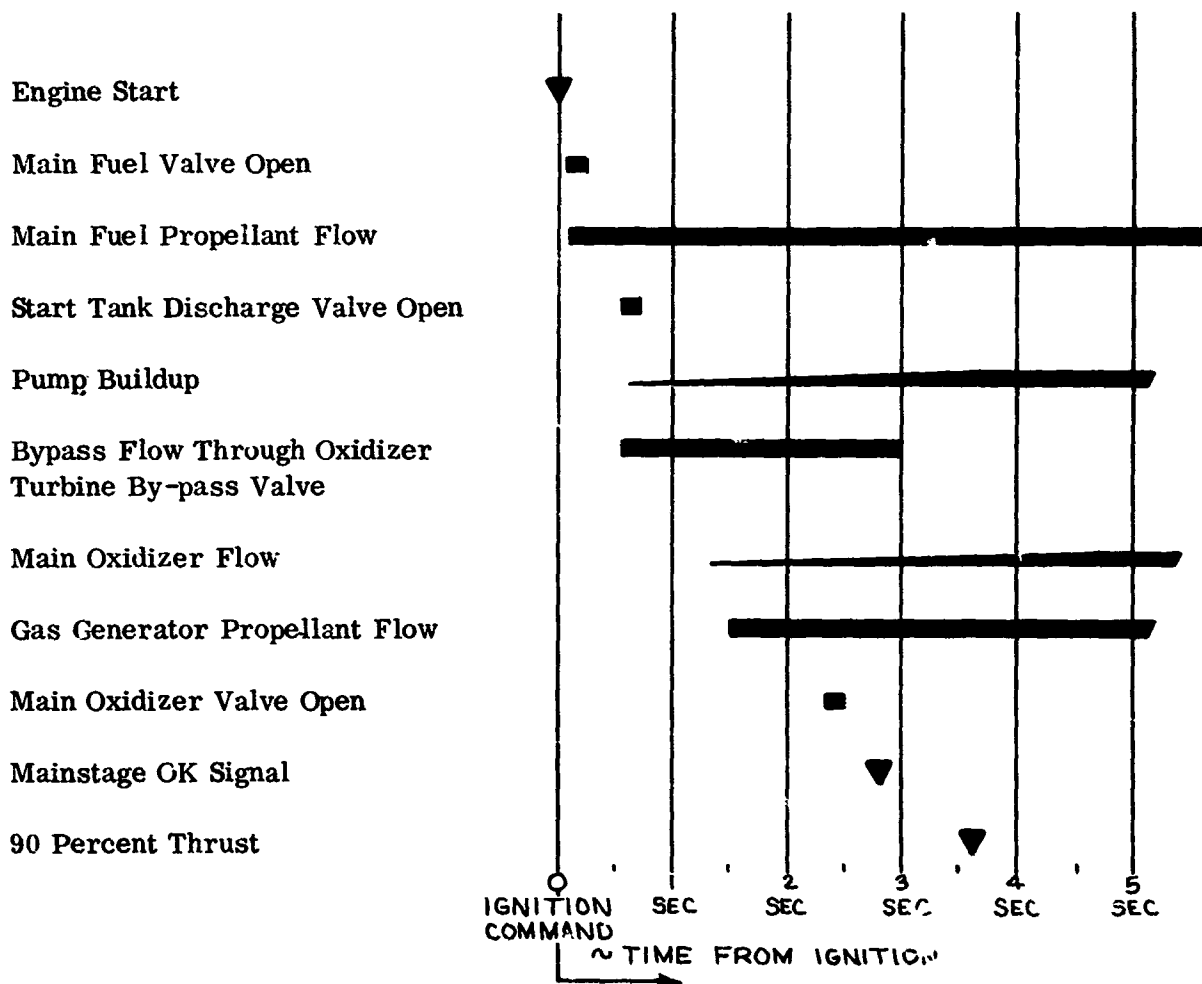
Figure 25

S-II Stage Configuration

## J-2 ENGINE OPERATION S-II STAGE

The operating cycle of the J-2 Engine consists of prestart, start, steady-state operation and cutoff sequences. During prestart, LOX and  $\text{LH}_2$  flow through the engine to temperature-condition the engine components, and to assure the presence of propellant in the turbopumps for starting. Following a timed cooldown period, the start signal is received by the sequence controller which energizes various control solenoid valves to open the propellant valves in the proper sequence. The sequence controller also energizes spark plugs in the gas generator and thrust chamber to ignite the propellant. In addition, the sequence controller releases  $\text{GH}_2$  from the start tank. The  $\text{GH}_2$  provides the initial drive for the turbopumps that deliver propellant to the gas generator and the engine. The propellant ignites, gas generator output accelerates the turbopumps, and engine thrust increases to main stage operation. At this time, the spark plugs are de-energized and the engine is in steady-state operation.

Steady-state operation is maintained until a cutoff signal is received by the sequence controller. The sequence controller de-energizes the solenoid valves which in turn close the engine propellant valves in the proper sequence. As a result, engine thrust decays and the cutoff sequence is complete.





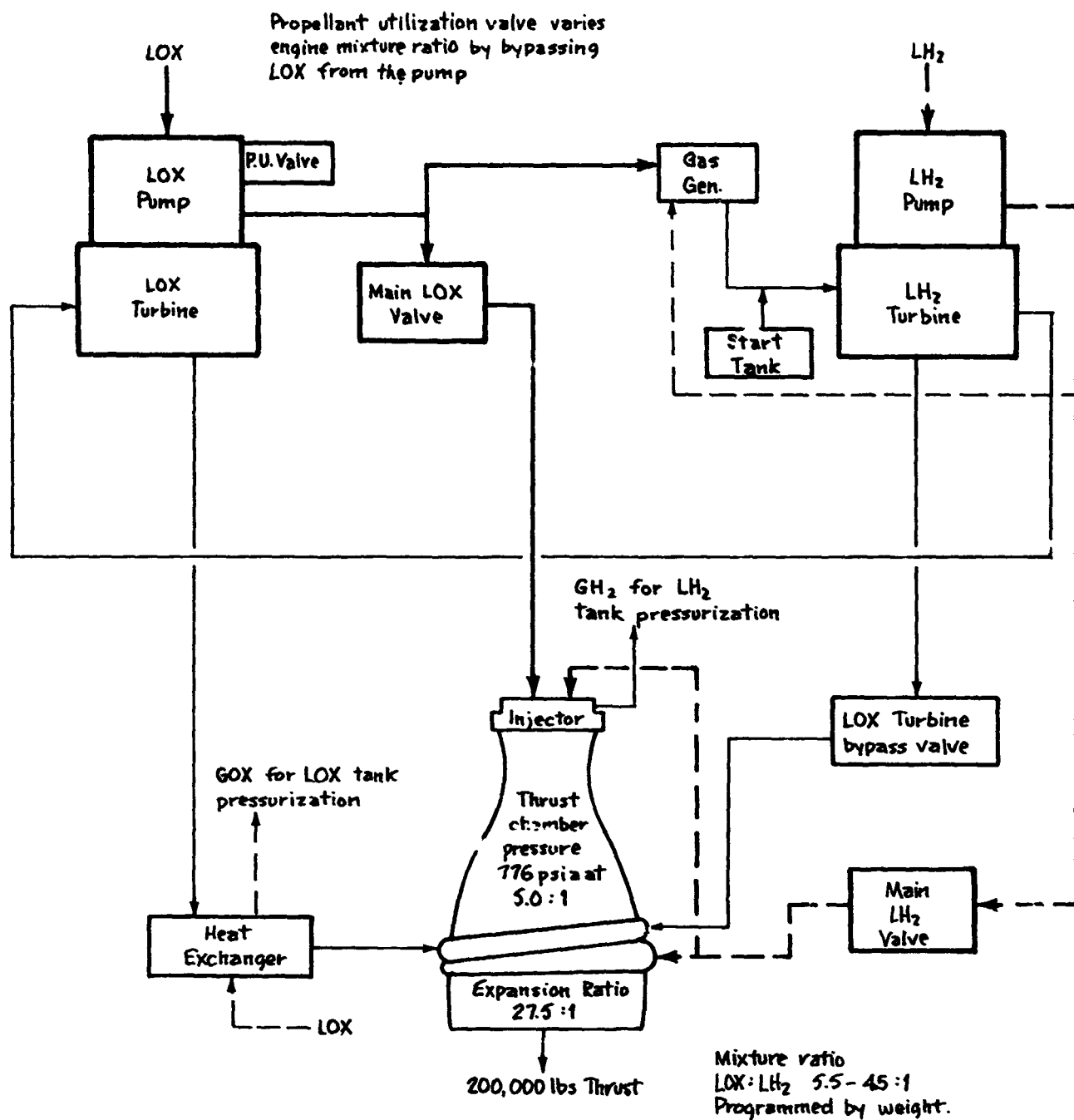


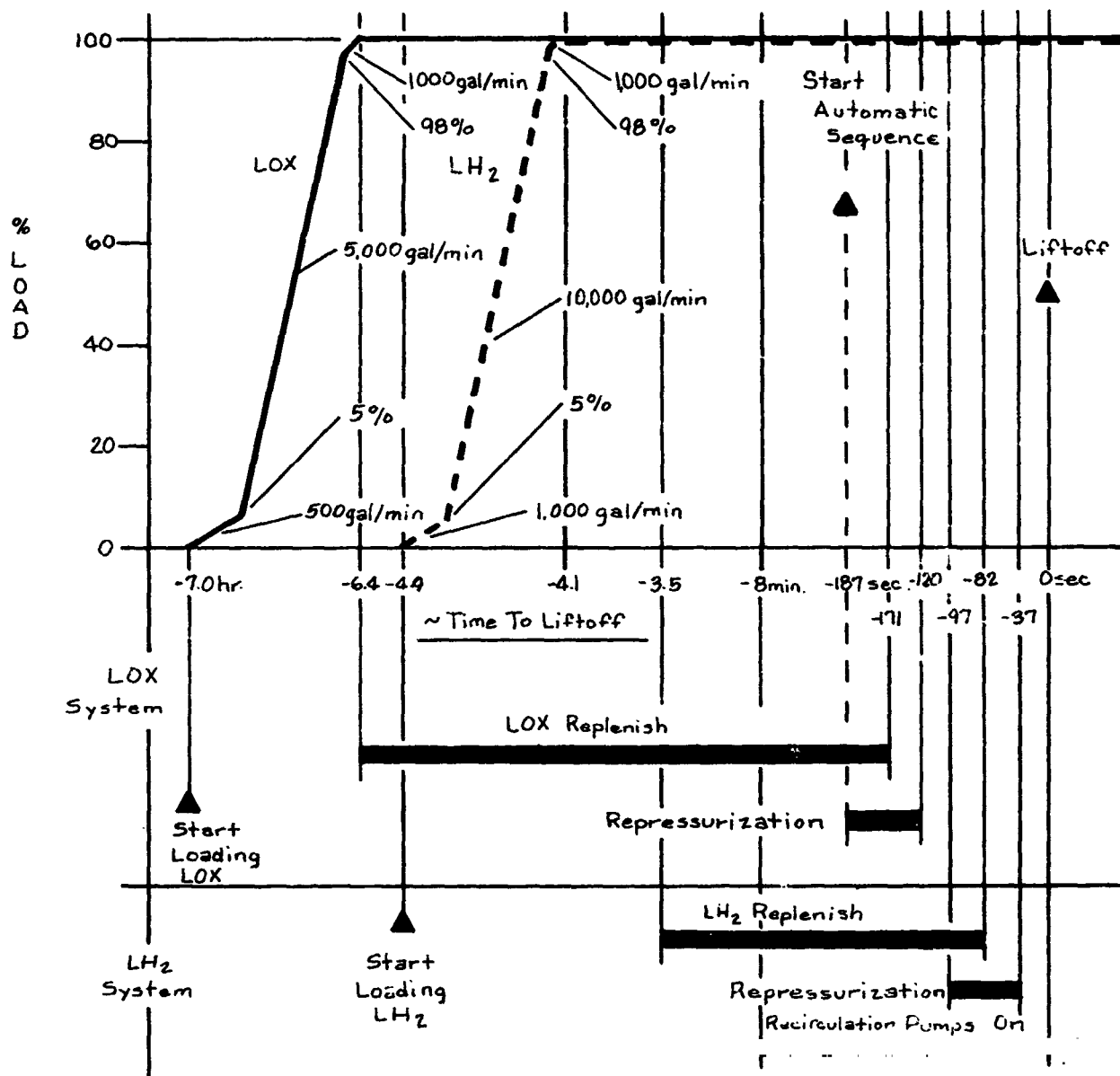
Figure 26

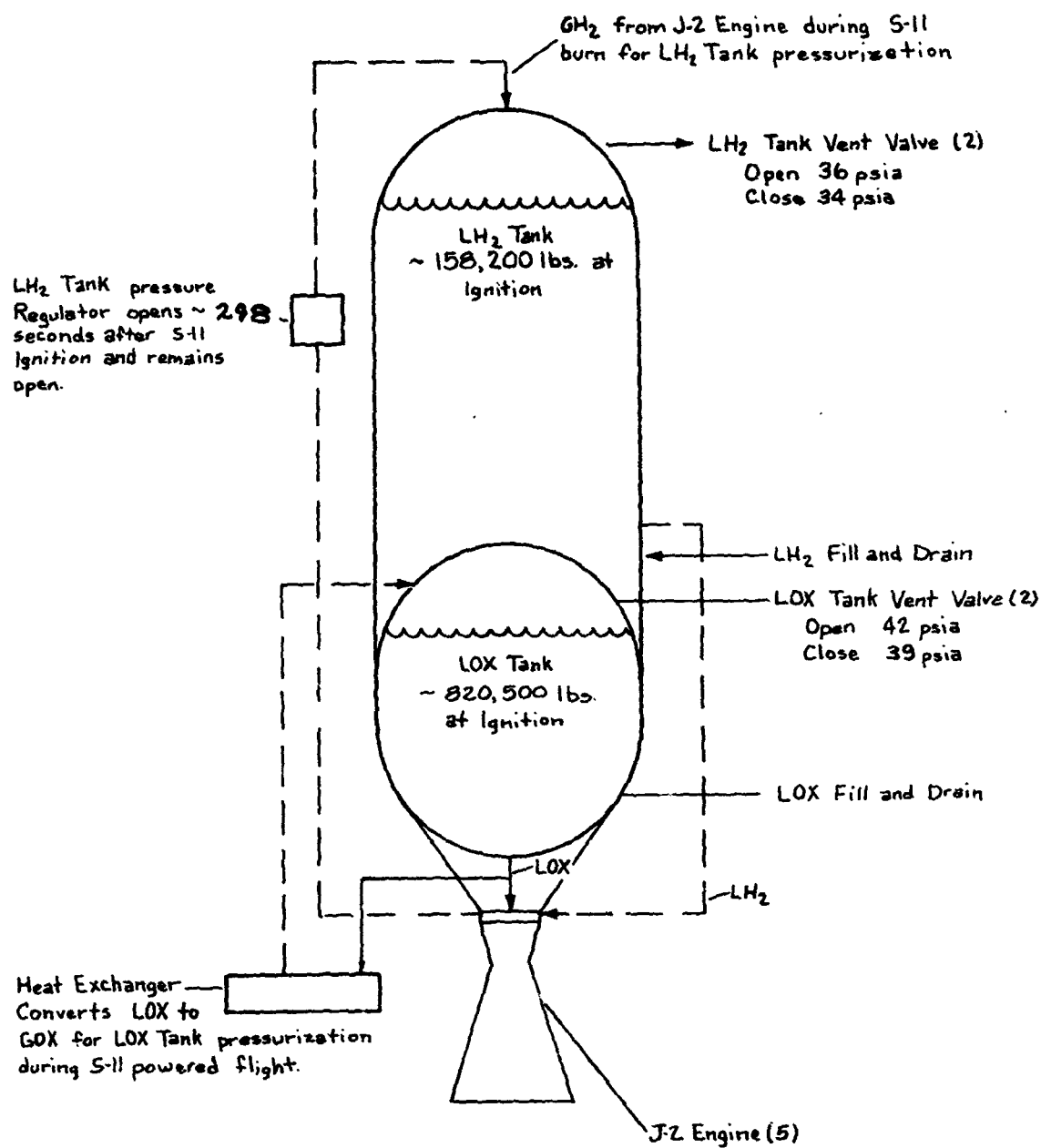
J-2 Engine System  
S-II Stage

## S-II STAGE PROPELLANT SYSTEM

The S-II Stage propellant system is composed of integral LOX/LH<sub>2</sub> tanks, propellant lines, control valves, vents, and prepressurization subsystems. Loading of propellant tanks and flow of propellants is controlled by the propellant utilization systems. The LOX/LH<sub>2</sub> tanks are prepressurized by ground source gaseous helium. During powered flight of the S-II Stage, the LOX tank is pressurized by GOX bleed from the LOX heat exchanger. The LH<sub>2</sub> tank is pressurized by GH<sub>2</sub> bleed from the thrust chamber hydrogen injector manifold; pressurization is maintained by the LH<sub>2</sub> Pressure Regulator.

## S-II PROPELLANT LOAD AND OPERATIONAL SEQUENCE





Total propellant at Ignition  
~ 978,800 lbs.  
Total propellant consumed after  
Ignition ~ 972,200 lbs.

Figure 27

S-II Stage Propellant System

## S-II STAGE PROPELLANT MANAGEMENT SYSTEM

The propellant management system provides a means of monitoring and controlling propellants during all phases of stage operation. Continuous capacitance probes and point level sensors in the LOX and LH<sub>2</sub> tanks monitor propellant mass.

During powered flight, the capacitance probes provide outputs used to operate the propellant utilization (PU) system to monitor the amount of LOX flowing to the J-2 engine and also to minimize propellant residuals at engine cutoff. At J-2 engine start, a mixture ratio of 5.0:1 is utilized. At PU actuation (5.5 seconds after engine start) the mixture ratio shifts to 5.5:1 for the first portion of the engine burn. During the last portion of the burn the mixture ratio shifts to 4.7:1 striving for simultaneous depletion of LOX and LH<sub>2</sub> for maximum stage performance. Engine cutoff is initiated when any two of the five capacitance probes in either propellant tank indicates dry.

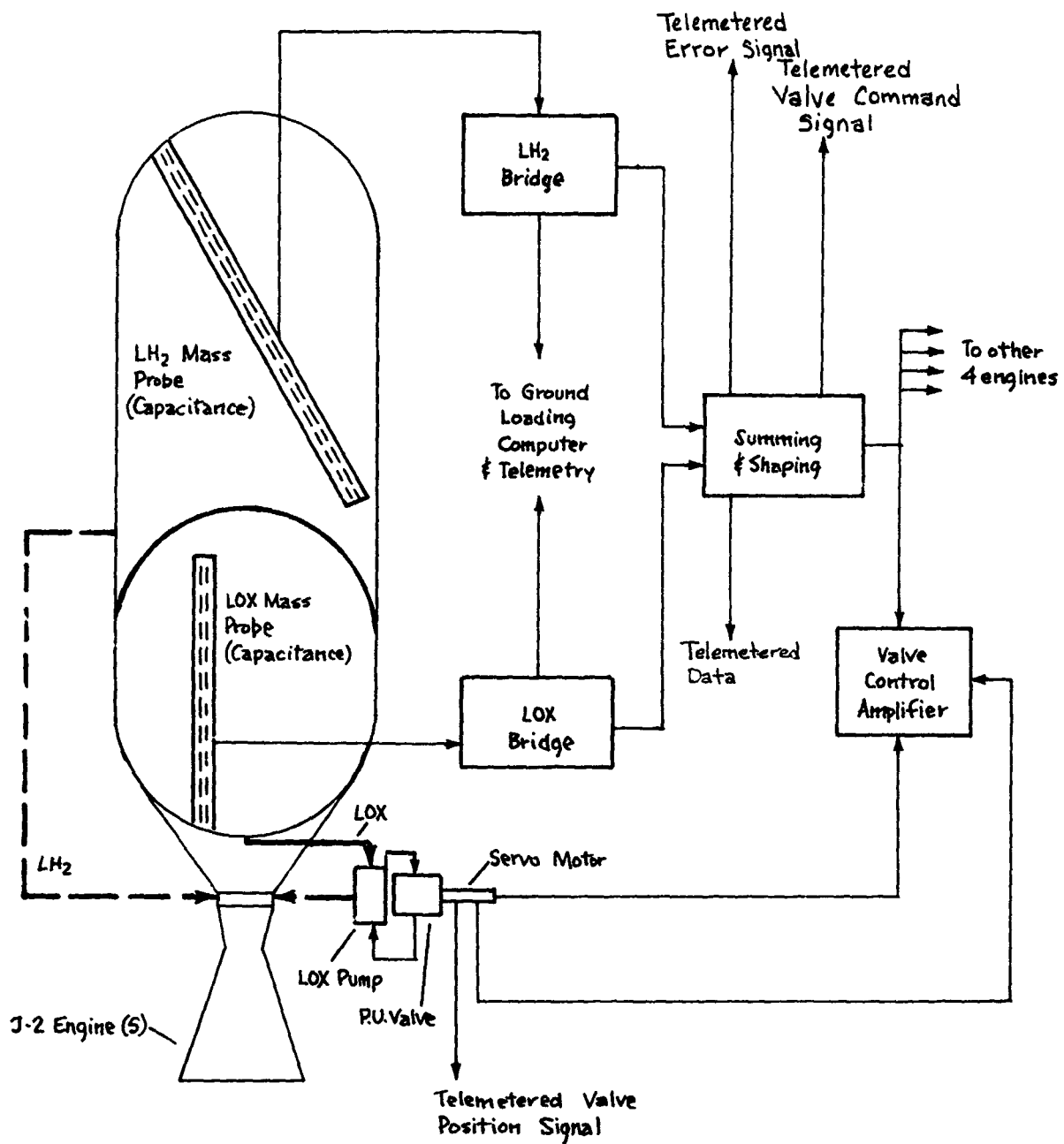


Figure 2B

S-II Stage Propellant Management System

## S-II STAGE THRUST VECTOR CONTROL SYSTEM

The four outboard engines are gimbal mounted to provide attitude control during powered flight. Attitude control is maintained by gimbaling one or more of the engines. Power for gimbaling is supplied by four independent engine mounted hydraulic control systems.

Pitch, yaw, and roll control, during powered flight, is maintained by actuator control of the engine thrust vector.

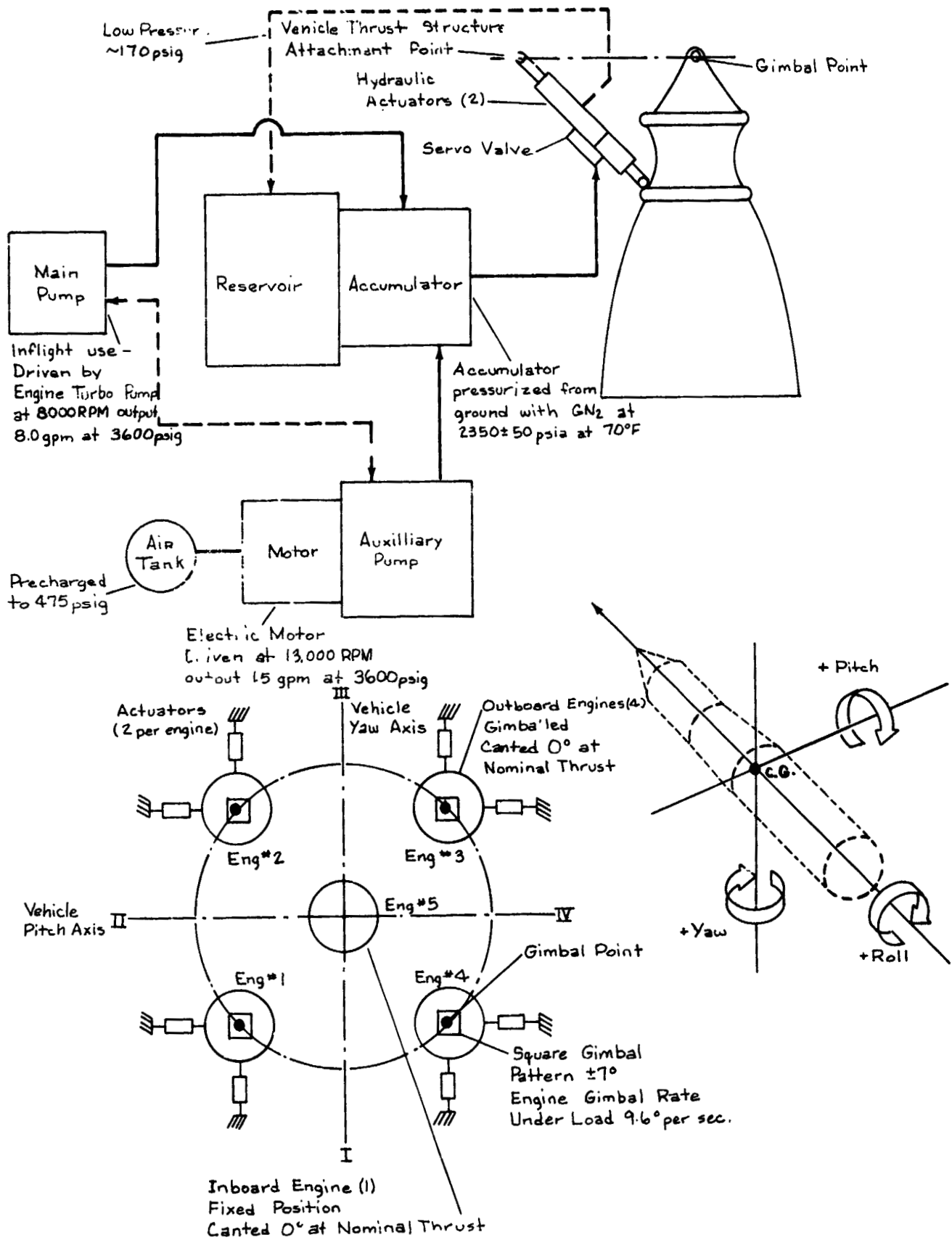


Figure 29

S-II Stage Thrust Vector Control System

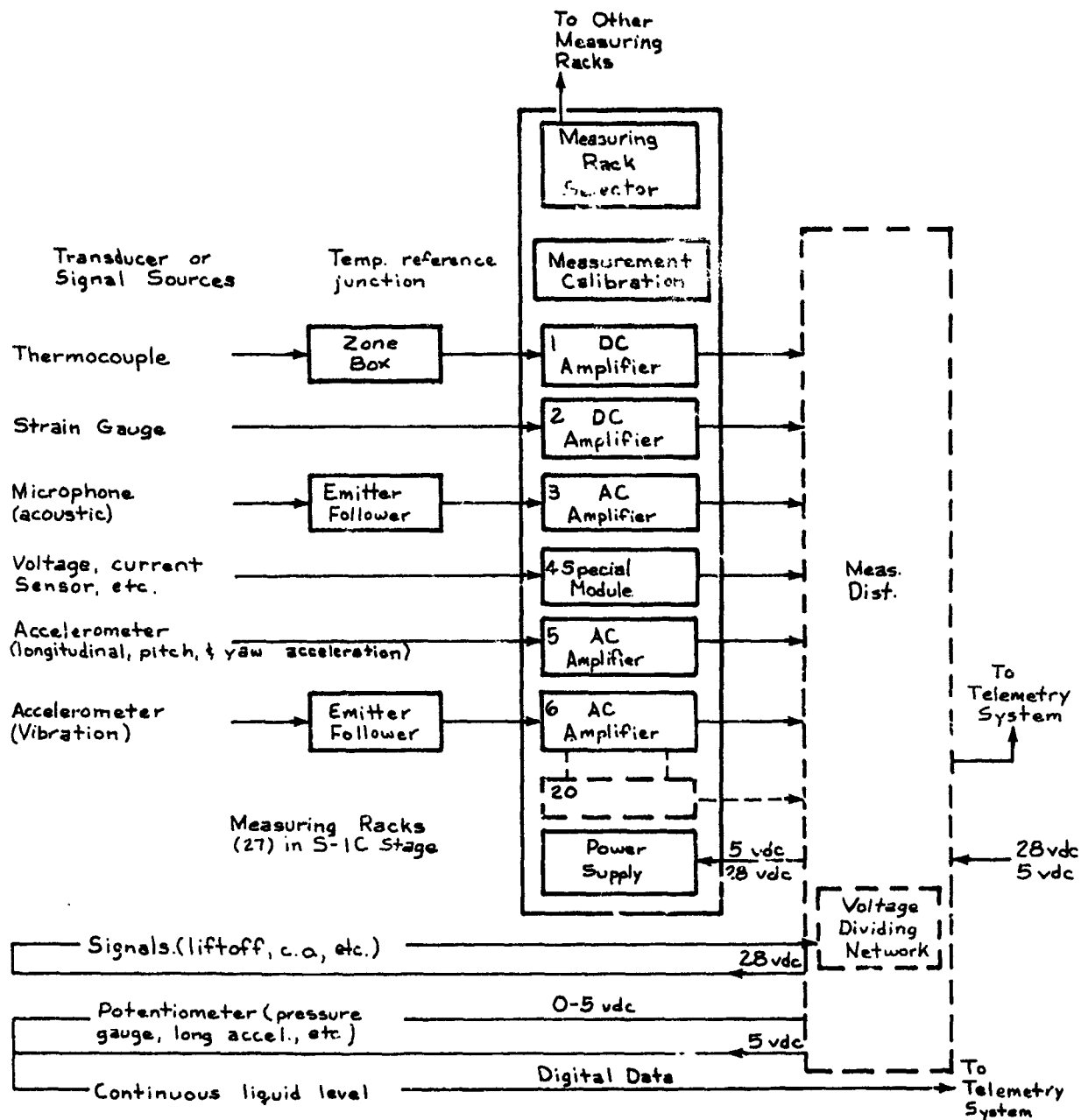
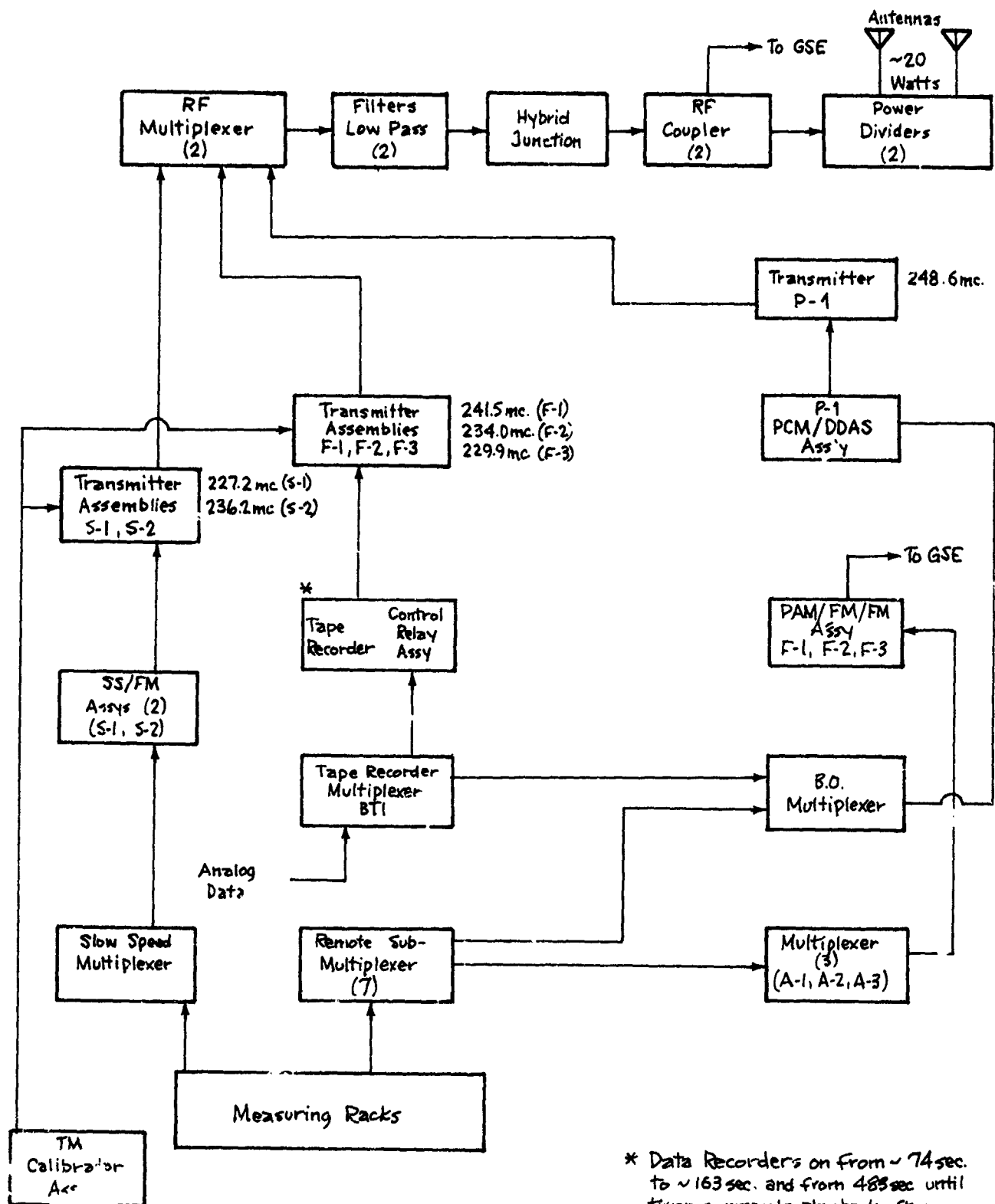


Figure 30

S-11 Stage  
Measuring System





\* Data Recorders on from ~ 74 sec. to ~ 163 sec. and from 483 sec until timer commands playback after separation

Figure 31

S-II Stage Telemetry System

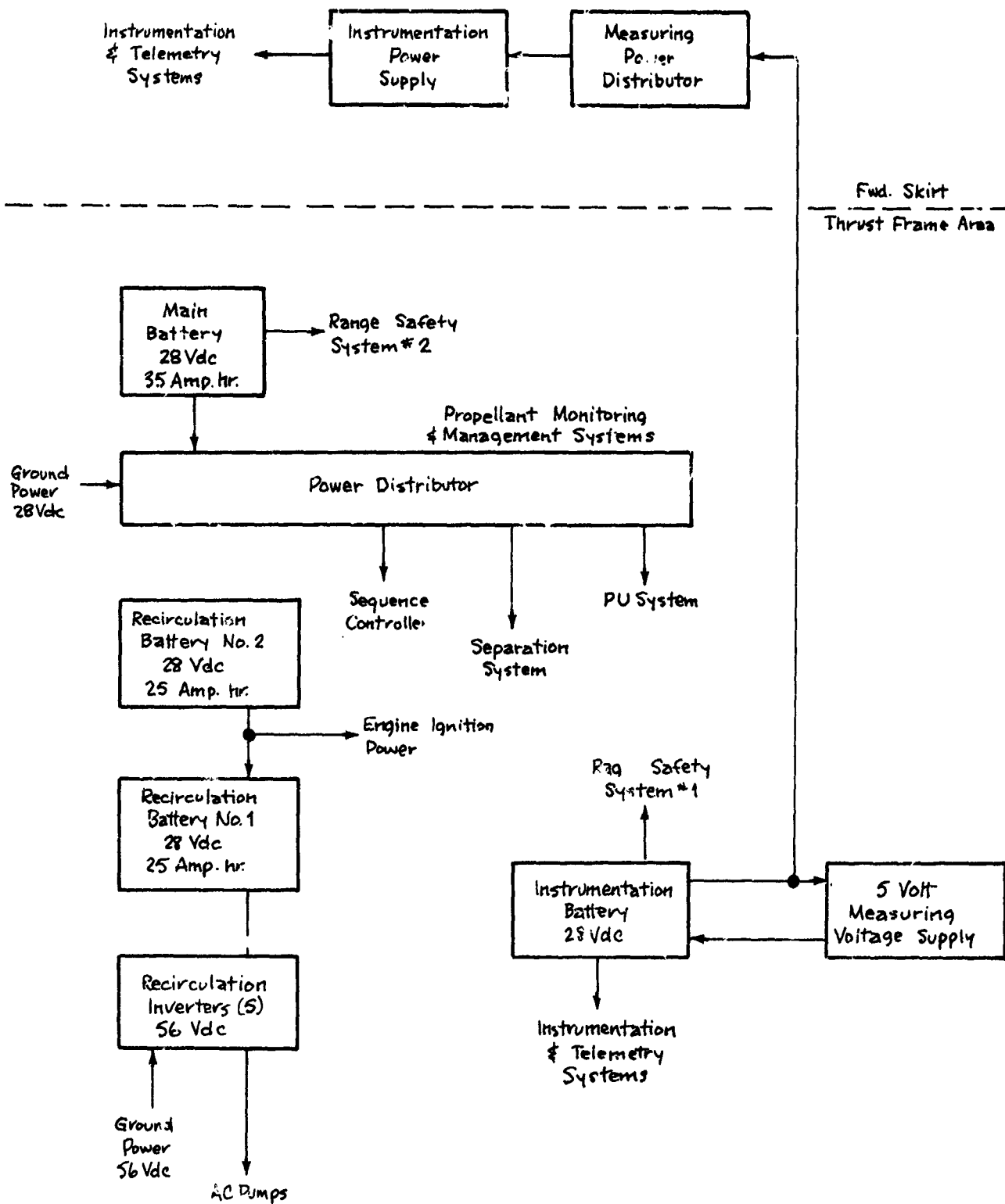


Figure 32

S-II Stage Electrical  
Power & Distribution System

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### **S-IVB STAGE**

The Saturn S-IVB is the third of the three booster stages. A single J-2 engine is designed to boost the payload into earth parking orbit during the first burn. A second stage burn is designed to provide vehicle position and velocity for lunar intercept.

The basic structural assembly of the S-IVB stage consists of; the forward skirt, propellant tanks, an aft skirt, thrust structure and aft interstage.

The two Auxiliary Propulsion System (APS) modules are located 180° apart on the aft skirt. Each module contains four engines; three 150-pound thrust and one 70 pound thrust. This APS system provides stage attitude control and main stage propellant control during coast flight.

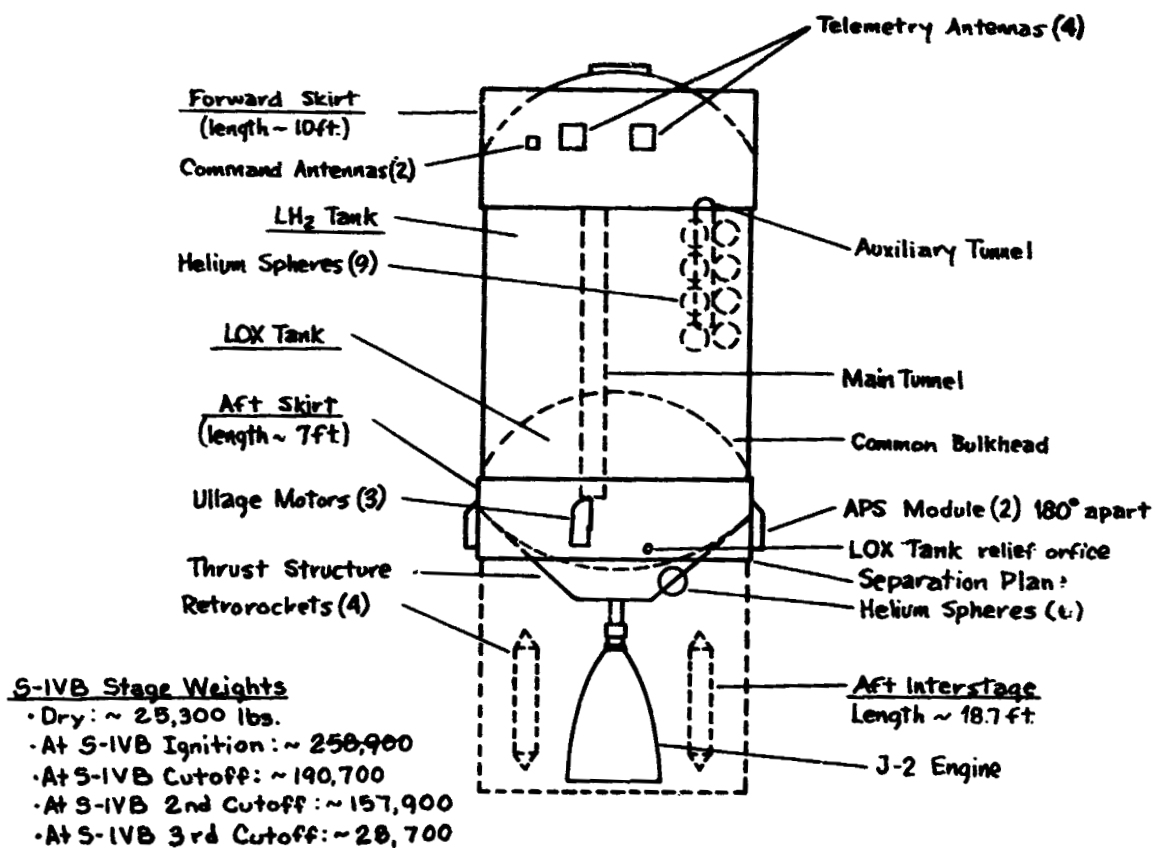
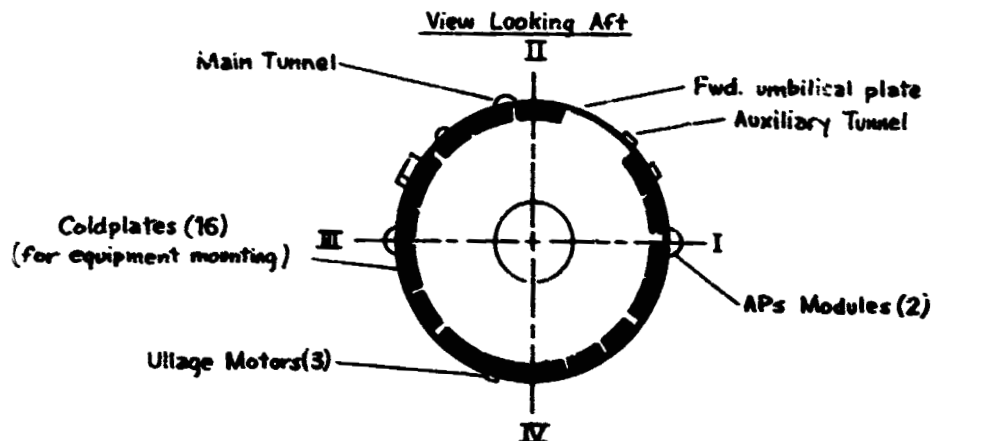


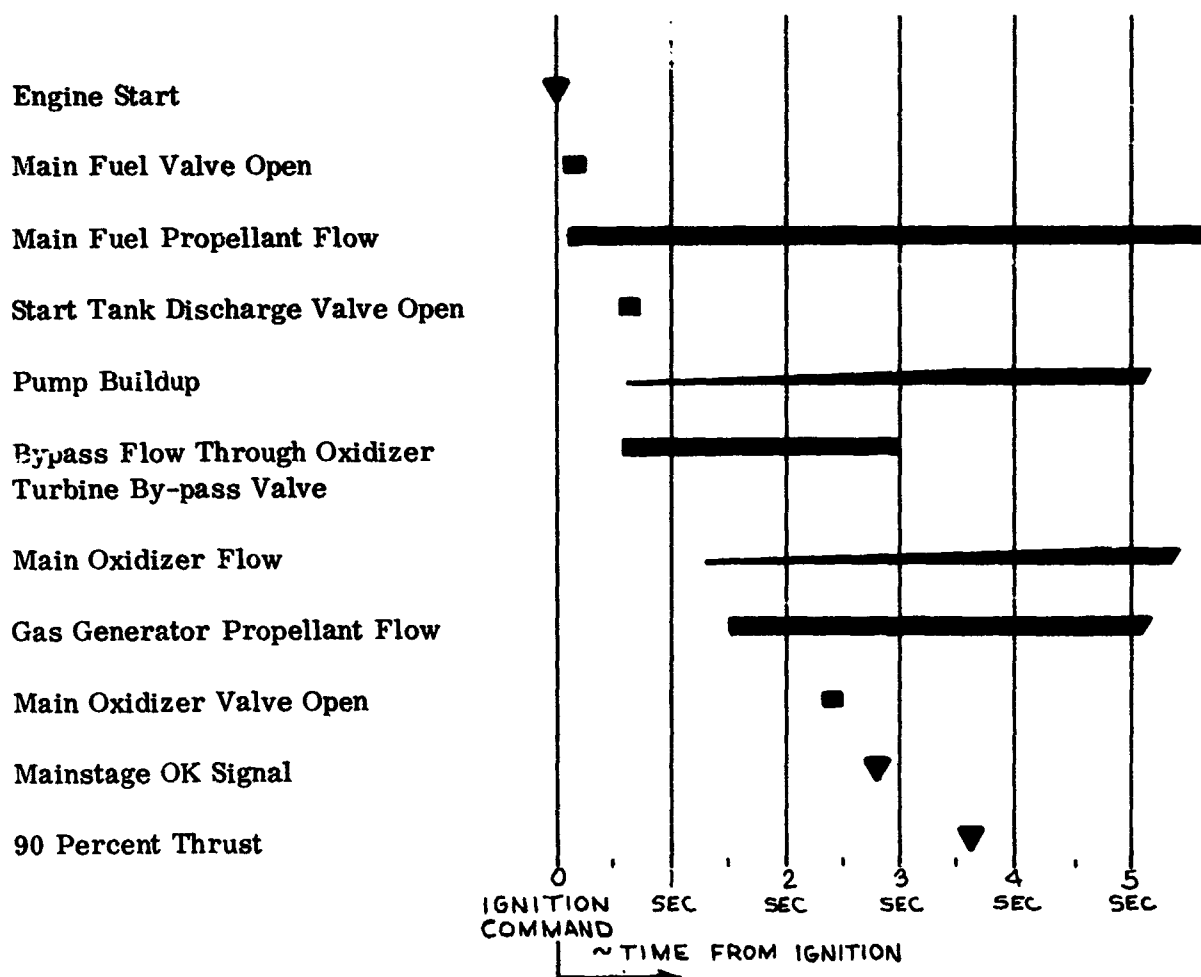
Figure 33

S-IVB Stage Configuration

## J-2 ENGINE OPERATION S-IVB STAGE

The operating cycle of the J-2 Engine consists of prestart, start, steady-state operation and cutoff sequences. During prestart, LOX and LH<sub>2</sub> flow through the engine to temperature-condition the engine components, and to assure the presence of propellant in the turbopumps for starting. Following a timed cooldown period, the start signal is received by the sequence controller which energizes various control solenoid valves to open the propellant valves in the proper sequence. The sequence controller also energizes spark plugs in the gas generator and thrust chamber to ignite the propellant. In addition, the sequence controller releases GH<sub>2</sub> from the start tank. The GH<sub>2</sub> provides the initial drive for the turbopumps that deliver propellant to the gas generator and the engine. The propellant ignites, gas generator output accelerates the turbopumps, and engine thrust increases to main stage operation. At this time, the spark plugs are de-energized and the engine is in steady-state operation.

Steady-state operation is maintained until a cutoff signal is received by the sequence controller. The sequence controller de-energizes the solenoid valves which in turn close the engine propellant valves in the proper sequence. As a result, engine thrust decays and the cutoff sequence is complete.



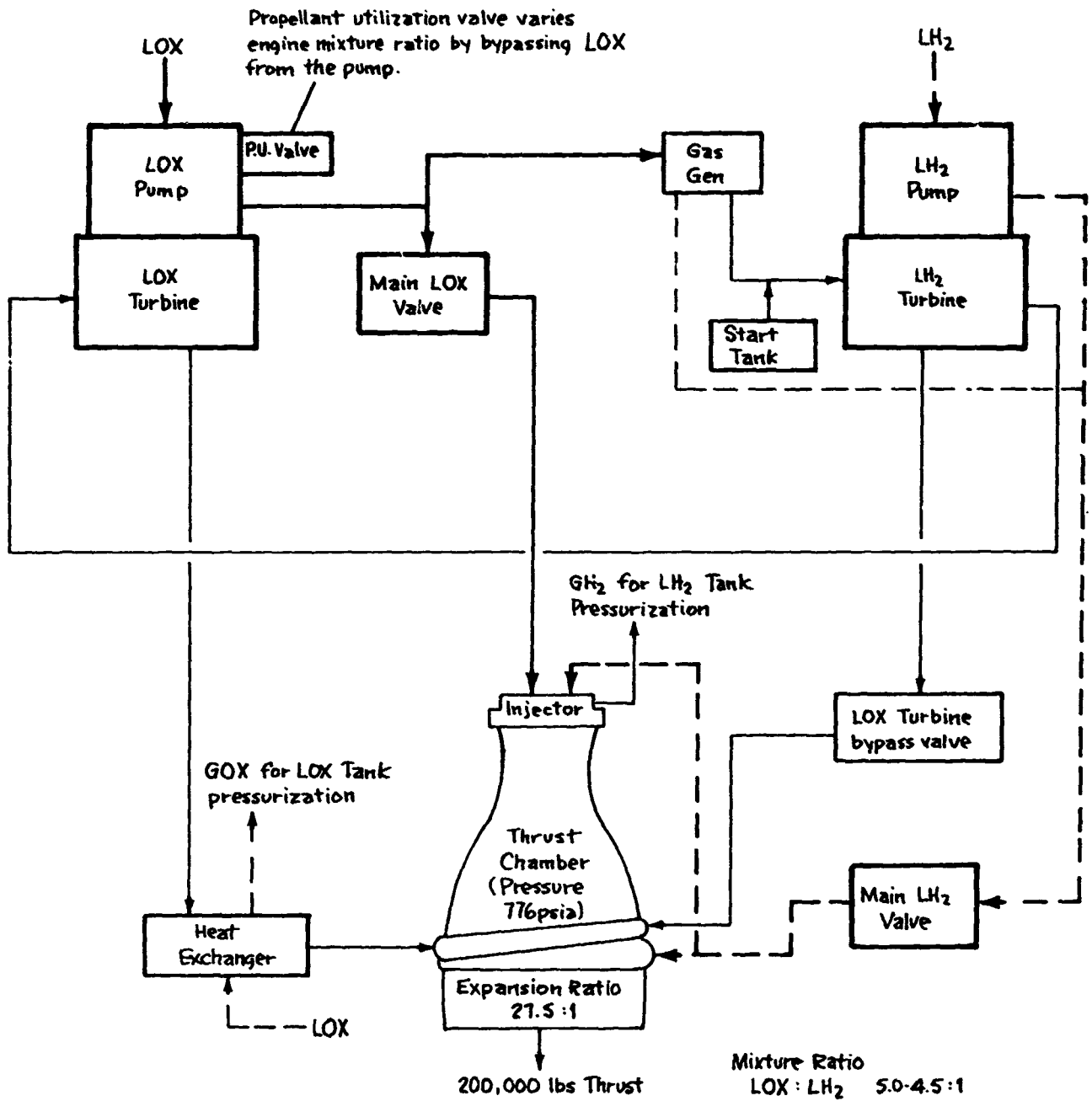


Figure 34

J-2 Engine System  
S-IVB Stage

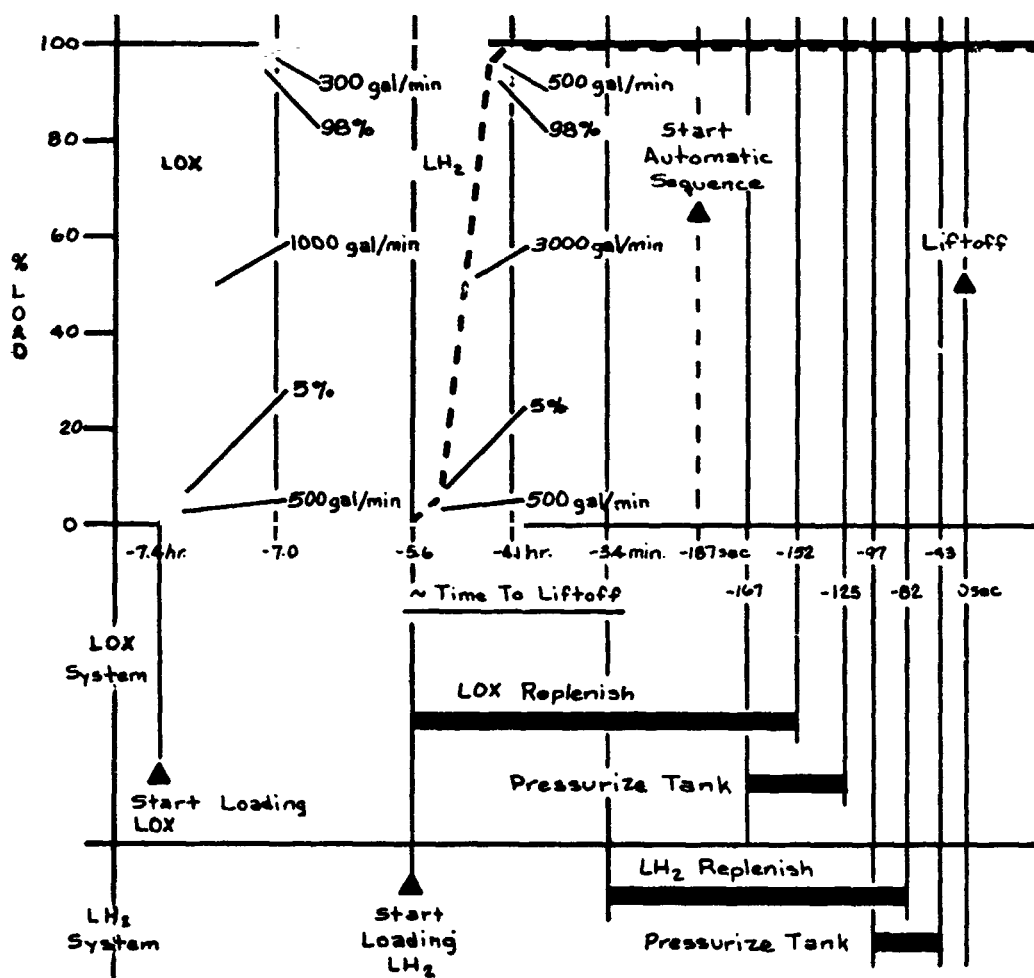
## S-IVB STAGE PROPELLANT SYSTEM

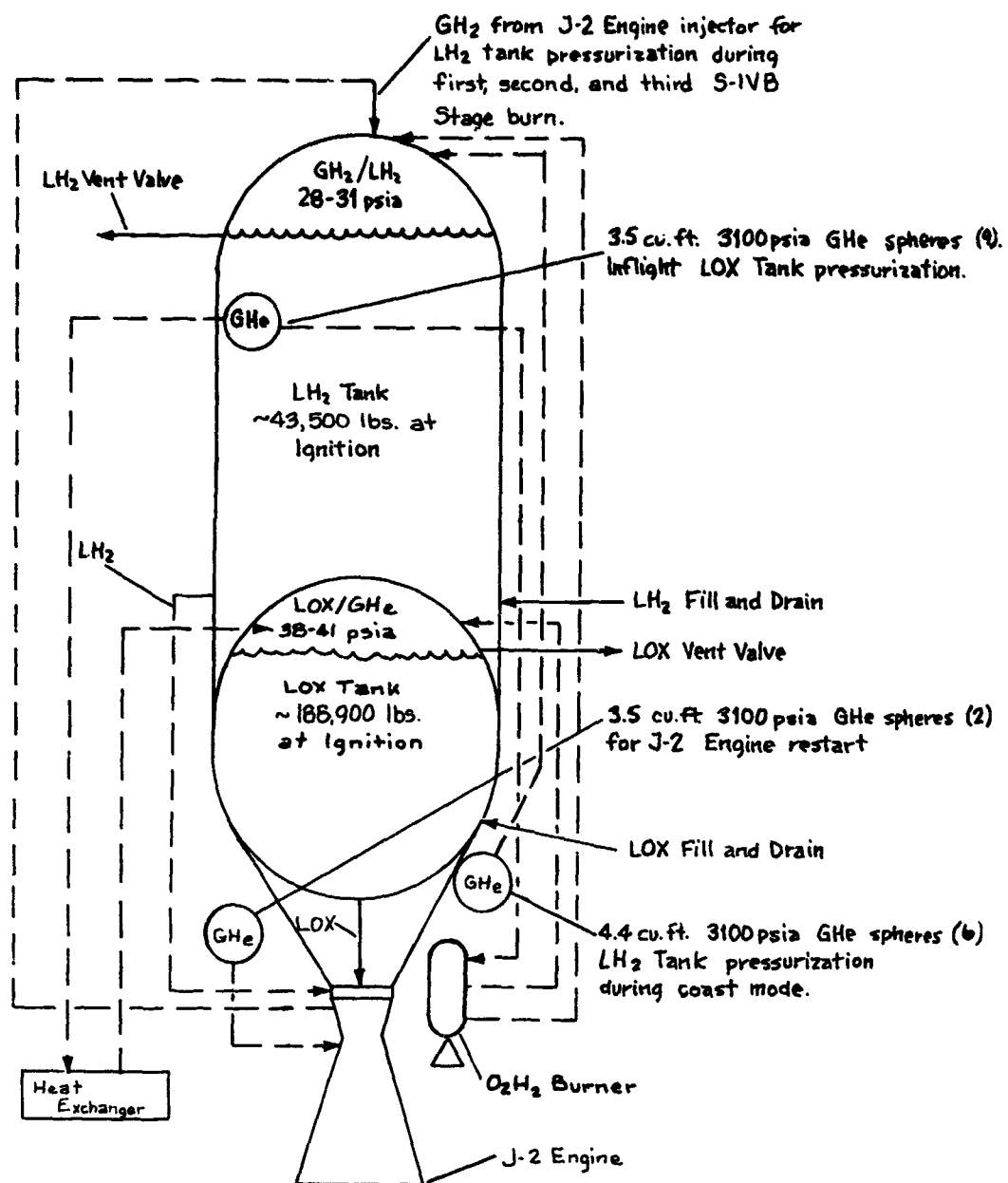
The S-IVB stage propellant system is composed of integral LOX/LH<sub>2</sub> tanks, propellant lines, control valves, vents and pressurization subsystems. Loading of the propellant tanks and flow of propellants is controlled by the propellant utilization system. Both propellant tanks are initially pressurized by ground source cold helium.

LOX tank pressurization during S-IVB stage burn is maintained by helium supplied from spheres in the LH<sub>2</sub> tank, which is expanded by passing through the heat exchanger, to maintain positive pressure across the common tank bulkhead and to satisfy engine net positive suction head. LH<sub>2</sub> tank pressurization during S-IVB stage burn is maintained by GH<sub>2</sub> from the J-2 engine injector. Pressurization of the LH<sub>2</sub> tank strengthens the stage in addition to satisfying engine net positive suction head.

Repressurization of the propellant tanks, prior to J-2 engine restarts, is attained by passing cold helium, from the helium spheres in the LH<sub>2</sub> tank, through the O<sub>2</sub>/H<sub>2</sub> burner. The heated helium is then routed to the propellant tanks. Should the O<sub>2</sub>/H<sub>2</sub> burner fail, ambient repressurization will ensure propellant tank pressure for engine restarts.

## S-IVB PROPELLANT LOAD AND OPERATIONAL SEQUENCE





Total propellant at ignition  
 ~ 232,400 lbs.  
 Total propellant consumed after  
 ignition ~ 230,700 lbs.

Figure 35

S-IVB Stage Propellant System



## S-IVB STAGE PROPELLANT MANAGEMENT SYSTEM

The propellant management system, in conjunction with the switch selector, controls mass propellant loading and engine mixture ratios (LOX to  $\text{LH}_2$ ) to ensure balanced consumption of LOX and  $\text{LH}_2$ .

Capacitance probes, mounted in the LOX and  $\text{LH}_2$  containers, monitor the mass of the propellants during powered flight. During flight, the LOX/ $\text{LH}_2$  capacitance probes are not utilized to control the propellant mixture ratio. This mode is considered to be an "open-loop", time-shift operation. During engine start and first burn, the ratio of LOX to  $\text{LH}_2$  is 5.0 to 1. The ratio at second and third engine restart is 4.5 to 1. Shortly after the engine reaches 90% thrust, the mixture ratio is shifted to 5.0 to 1 which will be used for the second burn (burn to intermediate orbit) and the third burn.

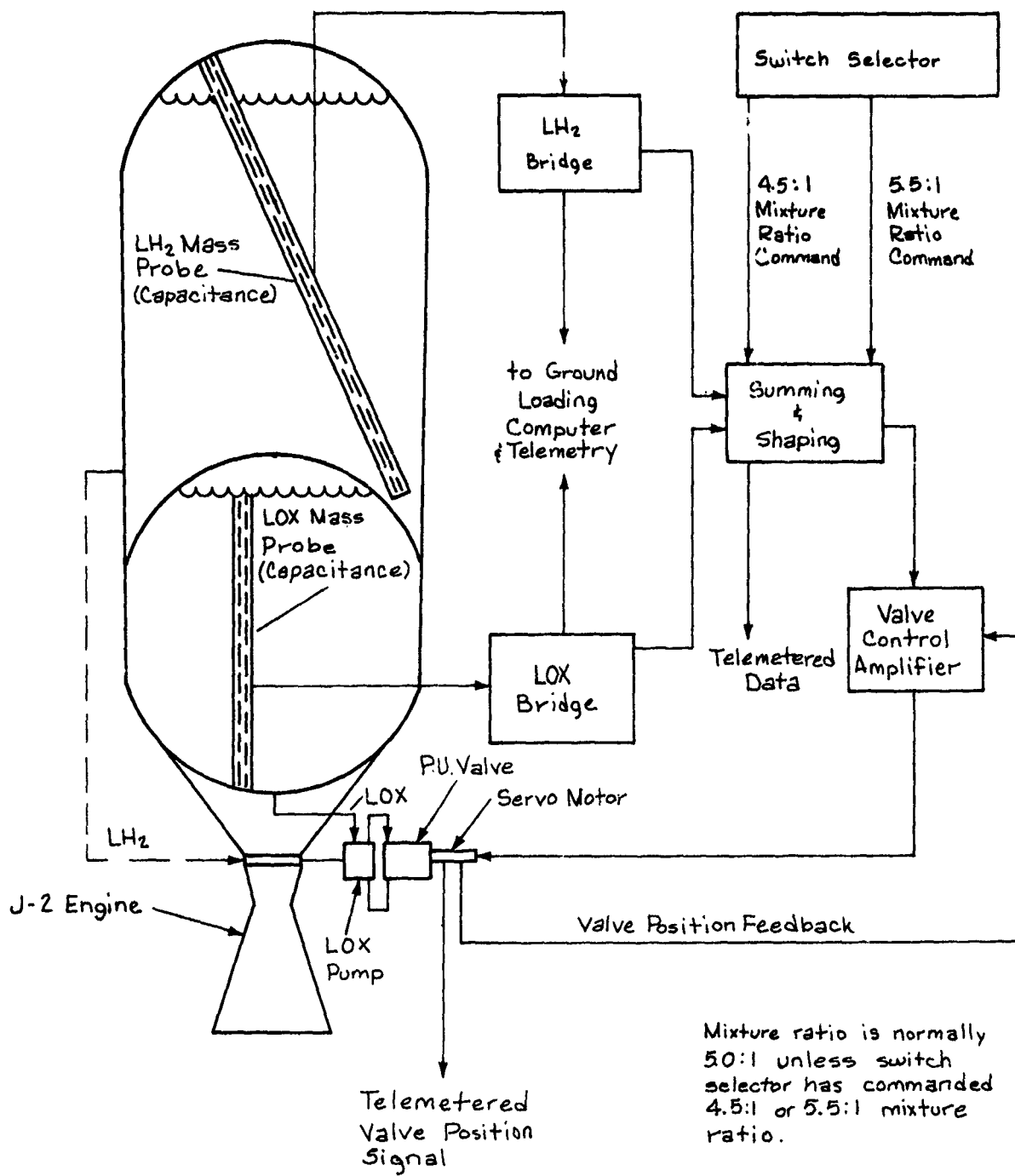


Figure 36

S-IVB Stage Propellant Management System

### S-IVB STAGE THRUST VECTOR CONTROL SYSTEM

The single J-2 Engine is gimbal mounted on the longitudinal axis of the S-IVB Stage. Power for gimbaling is supplied by a hydraulic control system mounted on the engine.

Pitch and yaw control, during powered flight, is maintained by actuator control of the engine thrust vector. Roll control of the stage is maintained by properly sequencing the pulse-fired hypergolic propellant thrust motors in the APS. When the stage enters the coast mode, the APS thrust motors control the stage in all three axes.

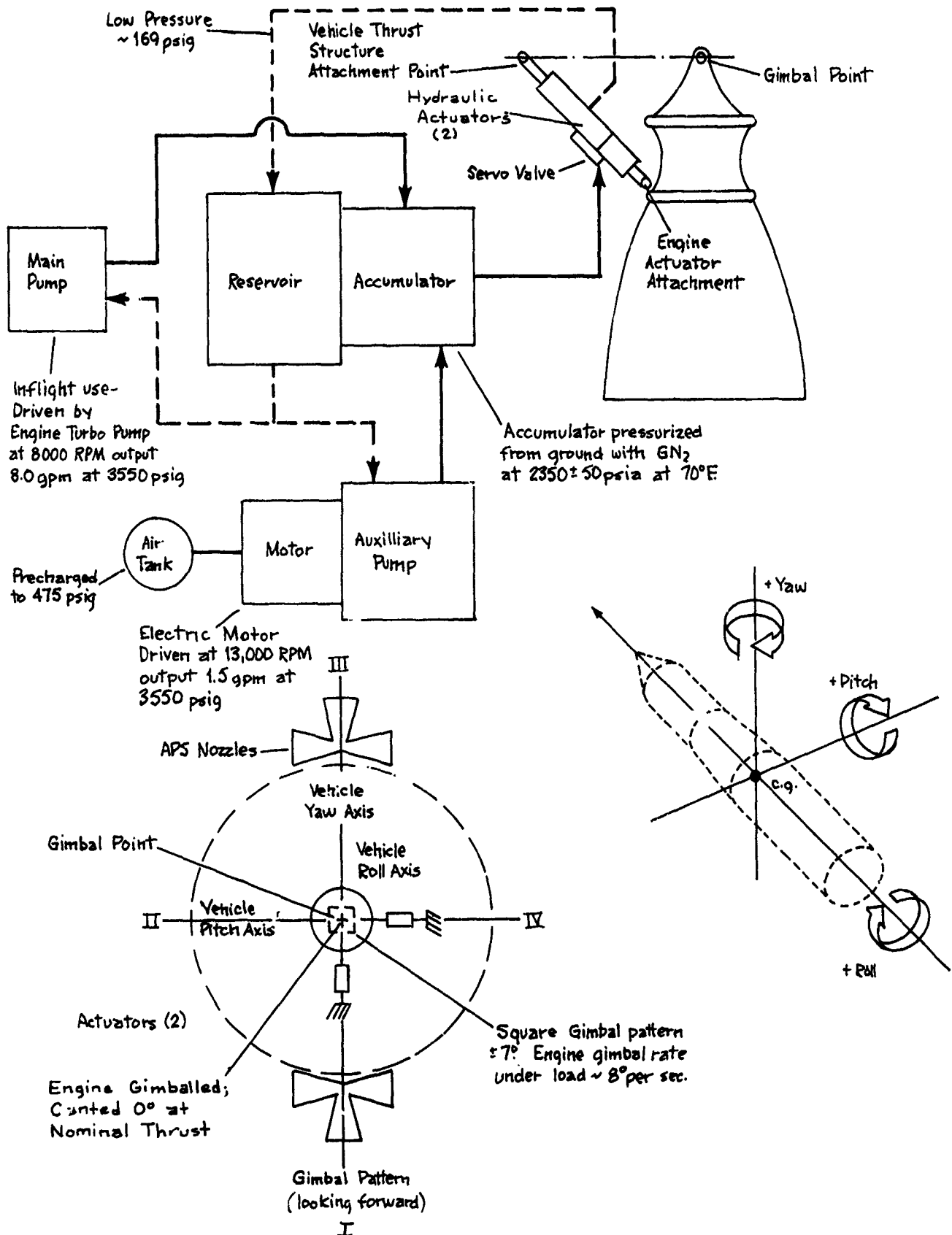


Figure 37

S-IVB Stage Thrust Vector Control System

## **AUXILIARY PROPULSION SYSTEM**

The APS consists of two self-contained attitude control modules mounted 180 degrees apart on the aft skirt of the S-IVB stage. Each attitude control module contains four thrust motors which use hypergolic propellant nitrogen tetroxide ( $N_2O_4$ ) and monomethylhydrazine (MMH) . The thrust motors are pulse-fired and no ignition system is required. Three thrust motors in each module provide pitch, yaw and roll control during the S-IVB coast mode of operation, and roll control during S-IVB powered flight. An ullaging engine is included in each module to settle propellants.

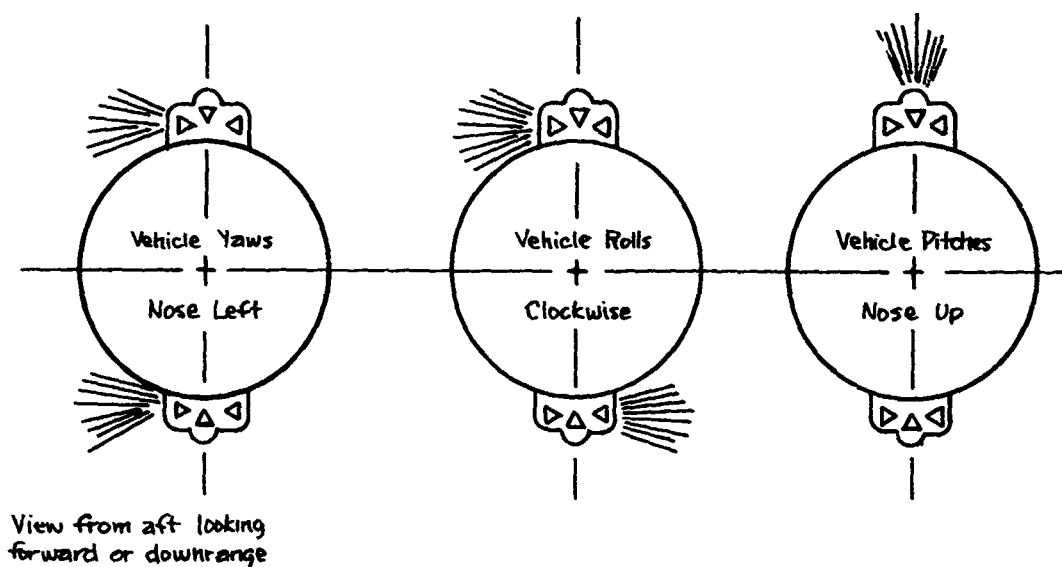
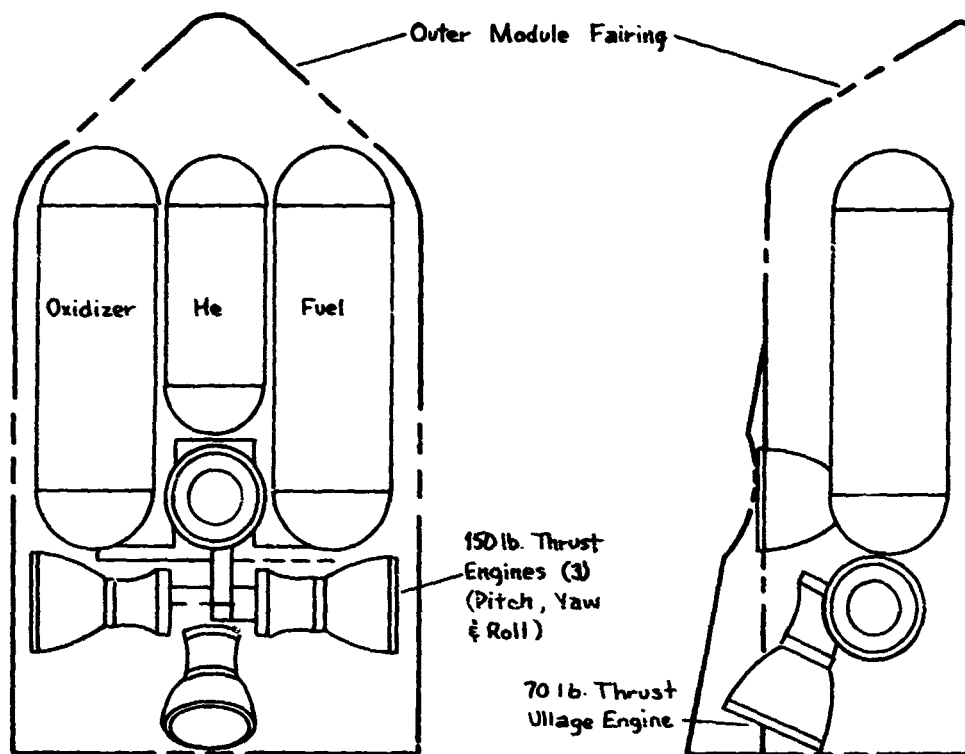


Figure 38

Auxiliary Propulsion System

# Transducers or Signal Sources

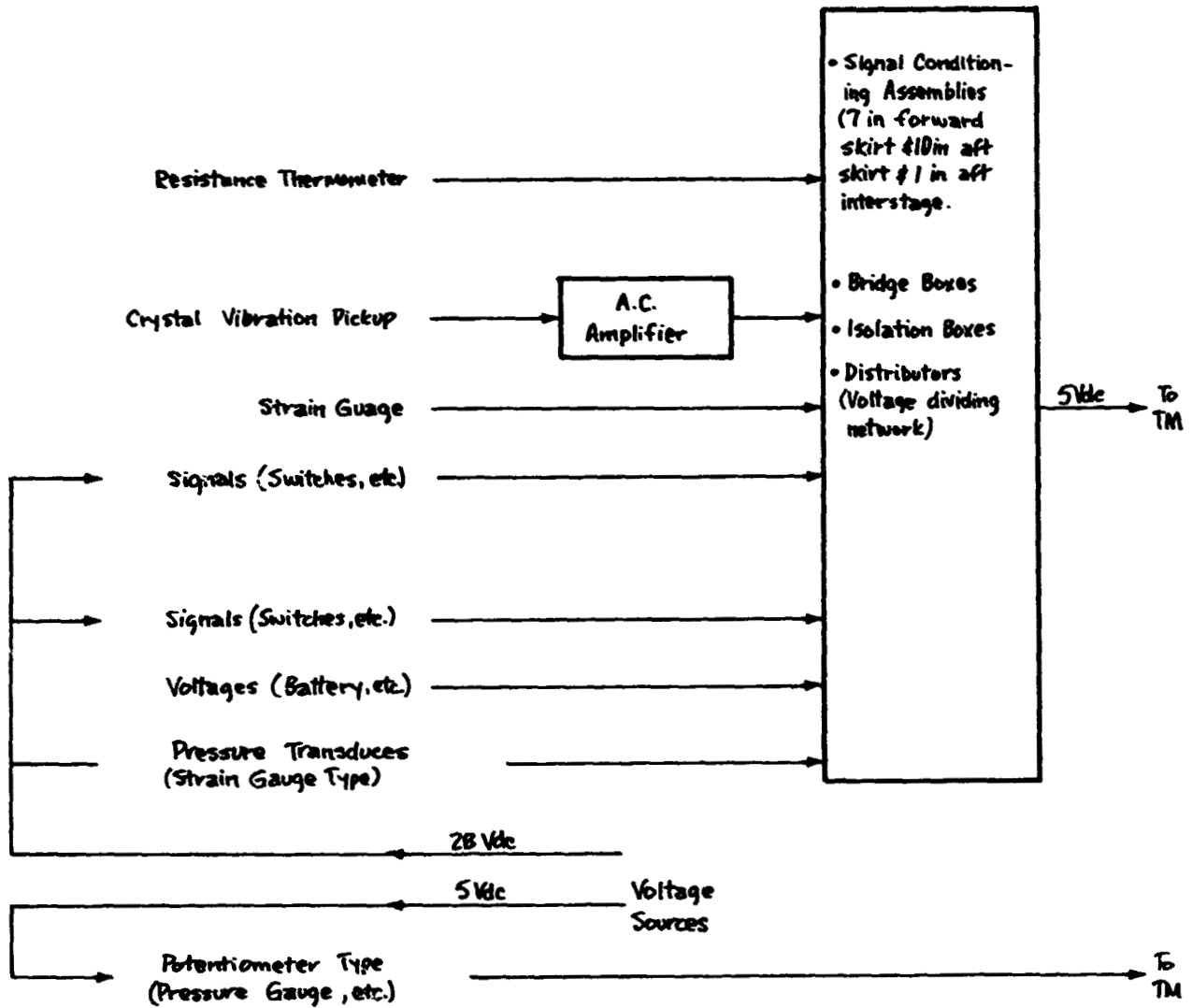


Figure 39

S-IVB Stage Measuring System

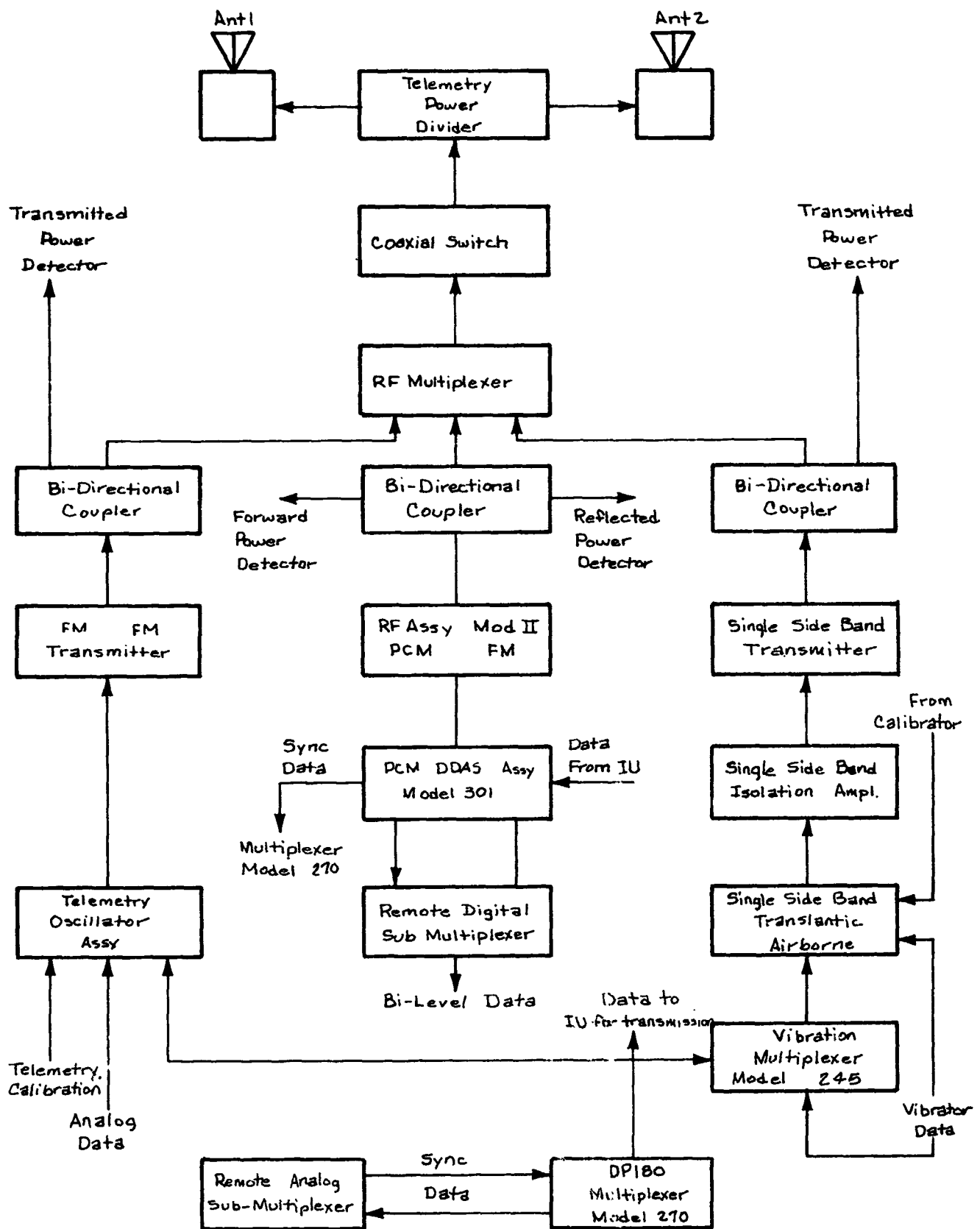


Figure 40

S-IVB Telemetry System



Note: Most forward Interstage Components  
Are Mounted On Coldplates

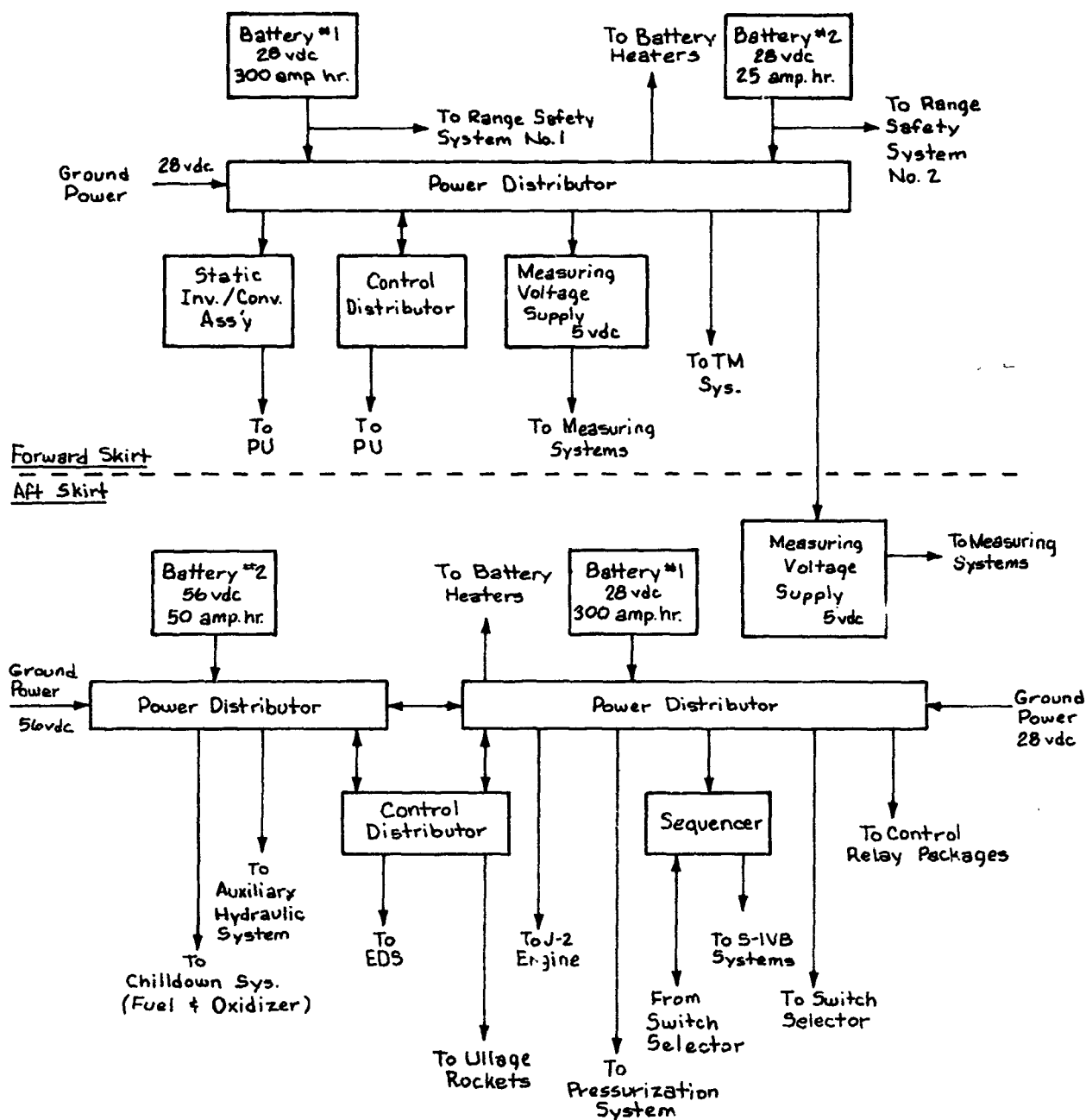


Figure 41

S-IVB Stage Electrical  
Power and Distribution System

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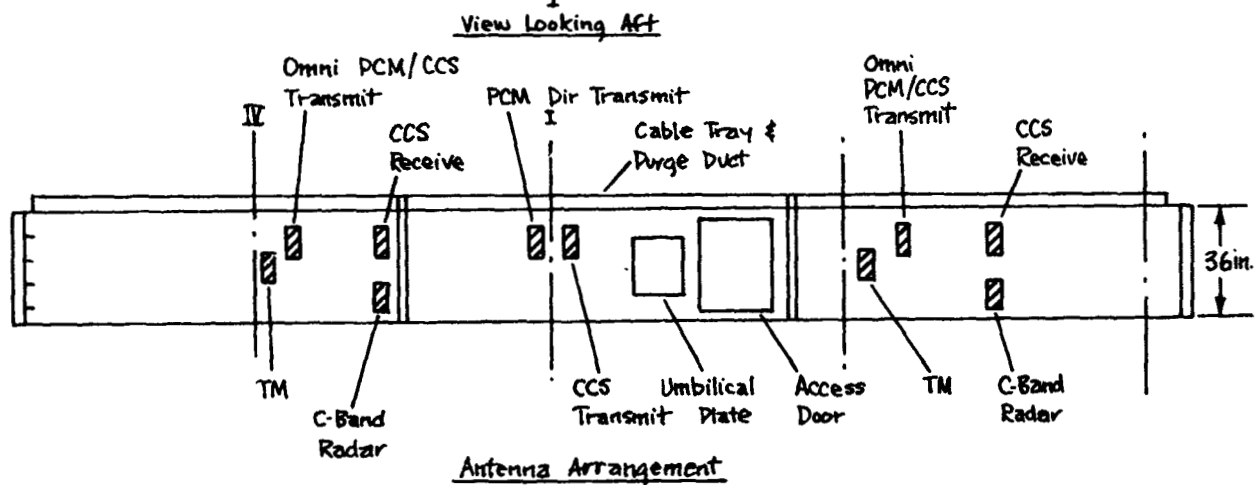
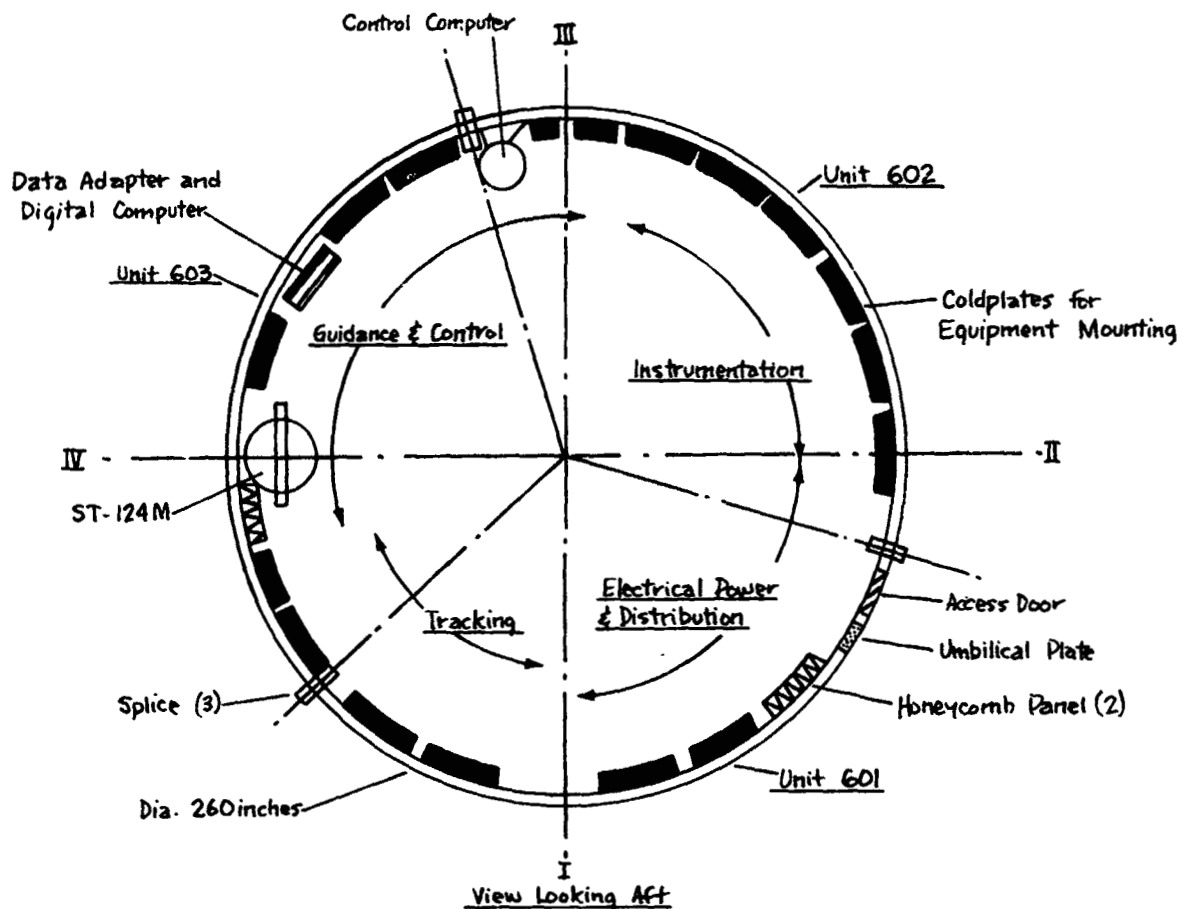
#### INSTRUMENT UNIT

The Instrument Unit is a cylindrical structure approximately 260 inches in diameter and 36 inches high which is attached to the forward end of the S-IVB stage.

The IU contains the guidance, navigation, and control equipment necessary for vehicle guidance through earth orbit and subsequent mission trajectory.

IU structure is composed of an aluminum alloy honeycomb sandwich material which was selected for its high strength-to-weight ratio, acoustical insulation, and thermal conductivity properties.

The cylinder is composed of three 120 degree segments -- the access door segment, the flight control computer segment, and the ST-124-M segment.



Weight:  
 • Dry ~ 4100 lbs.  
 Serviced ~ 4300 lbs.

Figure 42

Instrument Unit Configuration

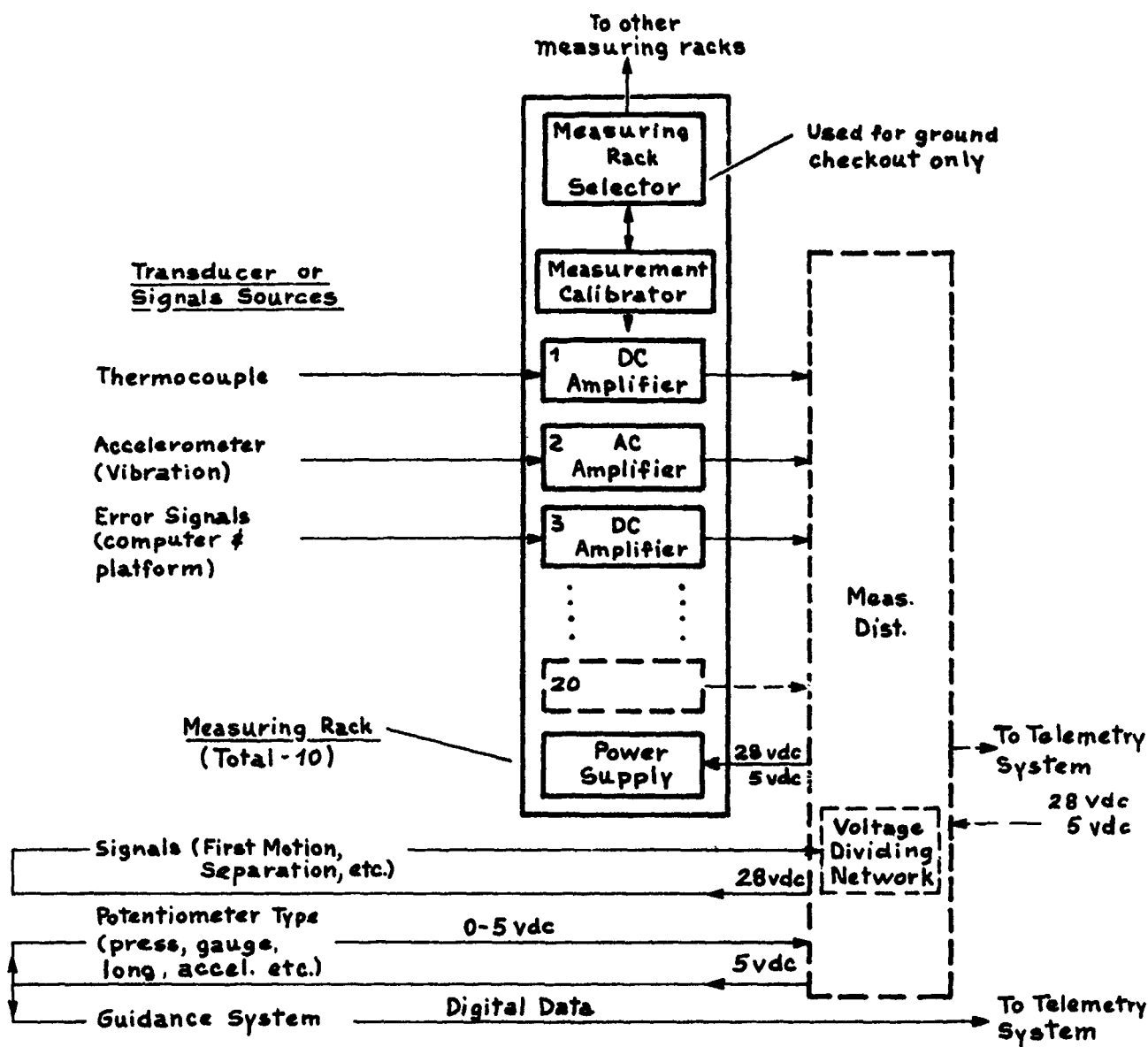


figure 43

Instrument Unit  
Measuring System

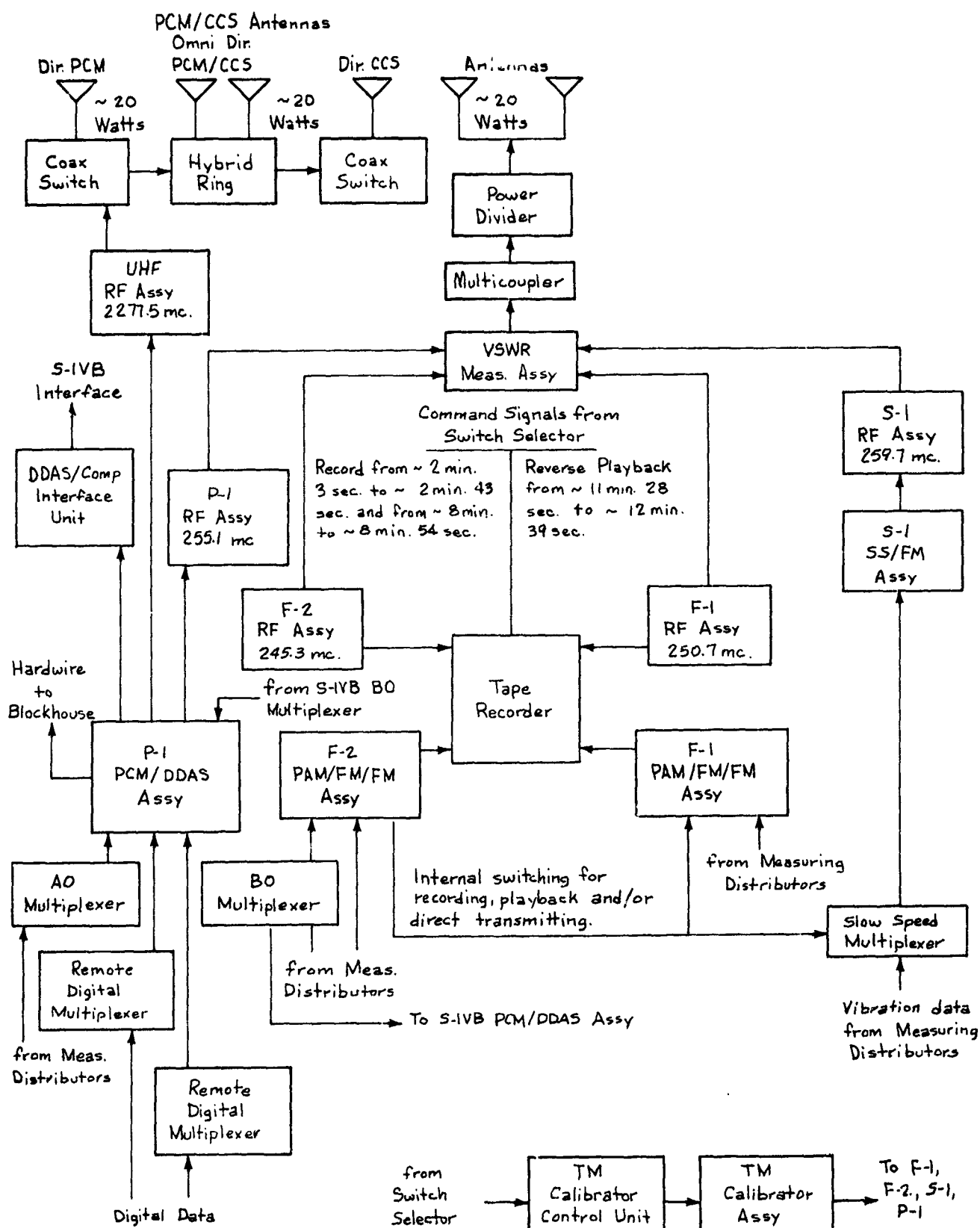


Figure 44

Instrument Unit  
Telemetry System

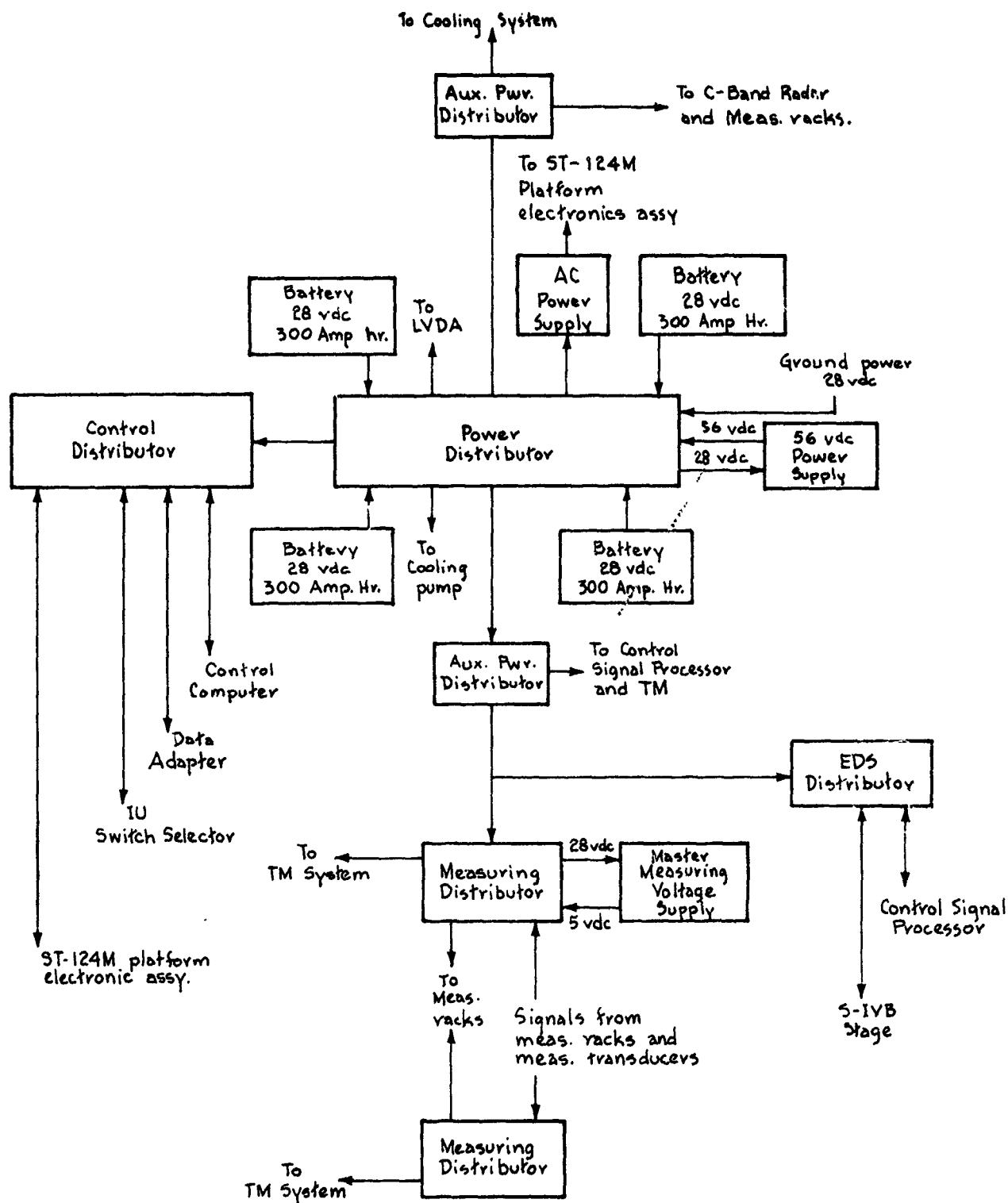


Figure 45

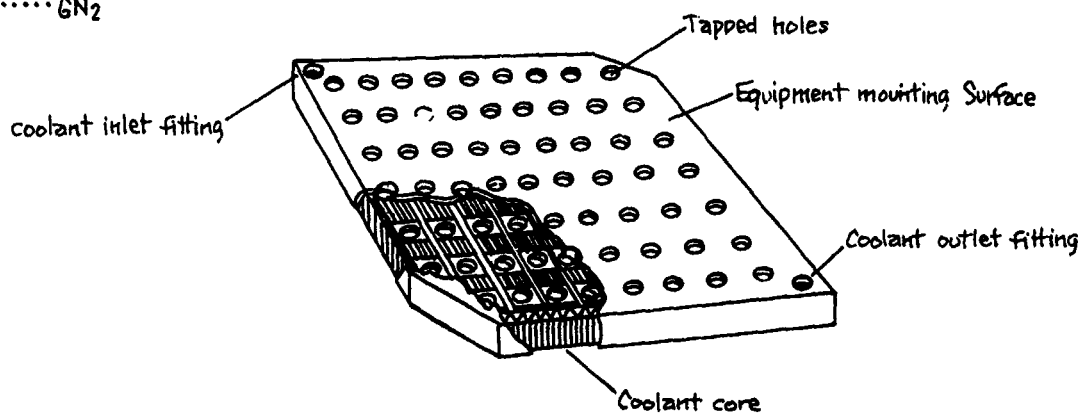
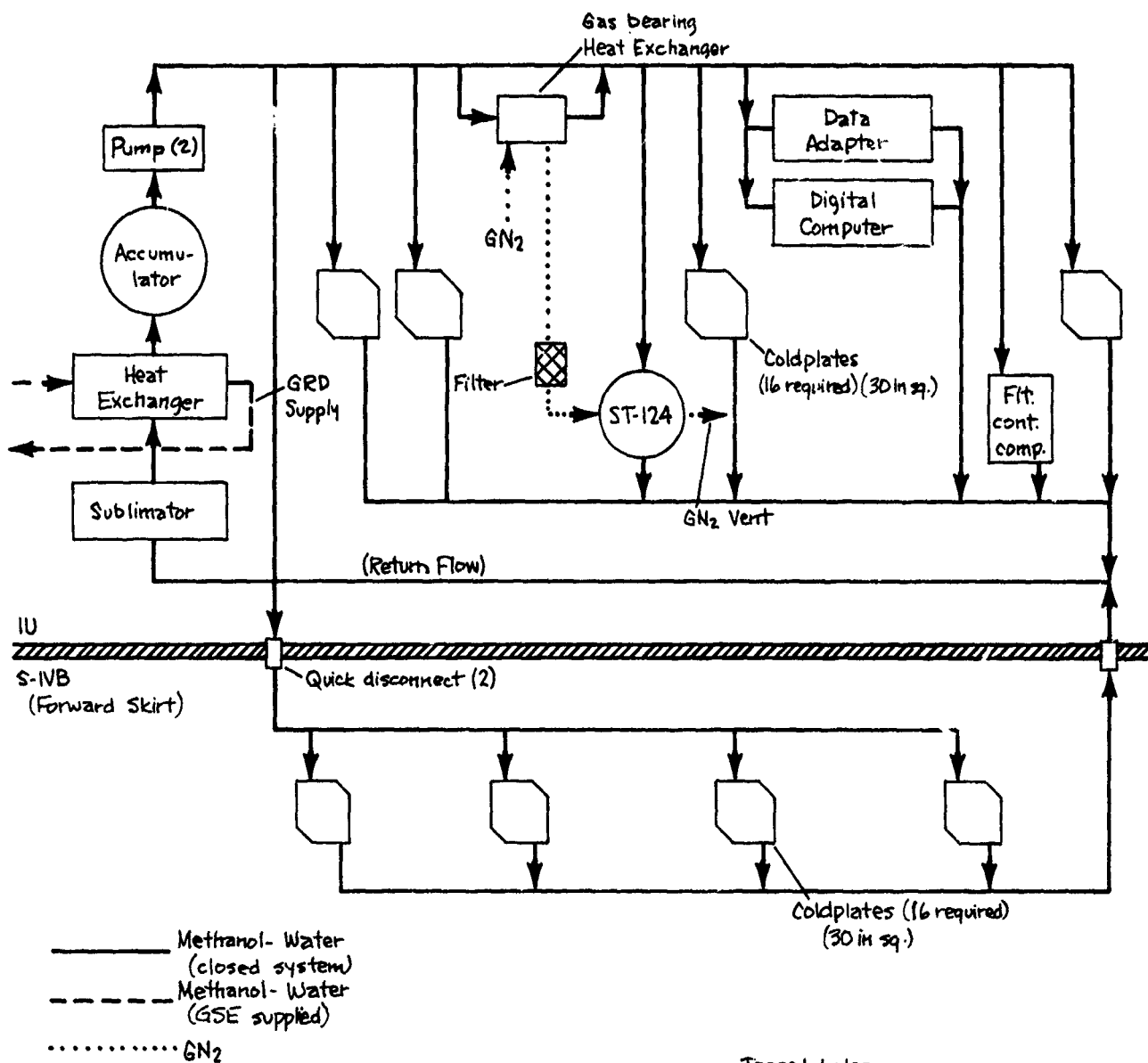
Instrument Unit Electrical  
Power and Distribution System

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ENVIRONMENTAL CONTROL SYSTEM (ECS)

Heat generated by the electronic equipment located in the S-IVB forward skirt and the IU is absorbed by circulating a methanol-water solution through the coldplate network.

Prior to liftoff, a temperature controlled methanol-water solution is supplied and circulated through the coldplates from the GSE. After liftoff the ECS is a selfcontained unit which begins operation 3 minutes after liftoff when the sublimator is activated.



Typical Coldplate

Figure 46

IU/S-IVB Environmental Control System



## SPACECRAFT DESCRIPTION

The Spacecraft for the AS-504 mission is composed of:

- Launch Escape System (LES)
- Command Module (CM)
- Service Module (SM)
- Lunar Module Adapter (LMA)
- Lunar Test Article (B) (LTA-B)

### Launch Escape System

The LES, which is jettisoned approximately 35 seconds after S-II Ignition, is made up of a Launch Escape Tower (LET), and a three-motor propulsion system (Tower Jettison, Launch Escape and Pitch Control Motors).

### Command Module

The Command Module for AS-504 is a Block II Configuration. The module's inner structure, or pressure vessel, is separated from the outer structure by a layer of insulation. A heat shield structure is made up in three segments consisting of a forward heat shield, a crew compartment heat shield, and an aft shield. The CM is slightly over 14 feet in length and is about 12 feet in diameter. A propulsion system consists of Reaction Control Engines which may operate pulsed or continuous.

### Service Module

The Service Module may be described as a cylindrical, aluminum, shell which is made up of honeycomb-sandwich panels and a forward and aft bulkhead. One gimbaled propulsion engine (capable of up to 30 restarts) and a reaction control system (4 clusters, 4 chambers each) make up the SM Propulsion System. The Command and Service Module are joined by 3 tension ties each of which is equipped with explosive charges for SM/CM separation.

### Lunar Module

The Lunar Module consists primarily of an Ascent and Descent Stage. The Ascent Stage, which contains the crew compartment, is equipped with a Reaction Control System which provides thrust capability, an ingress and egress hatch to the crew's compartment, VHF, S-Band and Rendezvous Radar capabilities plus numerous instrumentation and controls. The Descent Stage, consists primarily of a descent engine and four retractable landing gear assemblies. Over all weight of the Lunar Module is approximately 32,000 pounds.

During the AS-504 Mission, the prime test objectives relative to the Lunar Module will consist of; docking of the CSM/LM followed by LM extraction, a docked DPS burn, Extra Vehicular activity which includes astronaut travel (tethered) between LM and CSM, undocked DPS burn, LM staging, Ascent/CSM rendezvous and docking, and LM jettison.

Figure 47 illustrates the spacecraft configuration. Lunar Module is shown contained in the spacecraft Lunar Module Adapter (SLA). SLA panels are jettisoned at S-IVB-CSM separation.

Figure 51 is a detailed illustration of the Lunar Module.

Figure 52 illustrates the Lunar Module ascent and descent engine locations. Both will be tested during the mission.

Figure 53 contains a block diagram of the primary guidance and navigation system.

Figure 54 is a block diagram of the LM communications subsystem.

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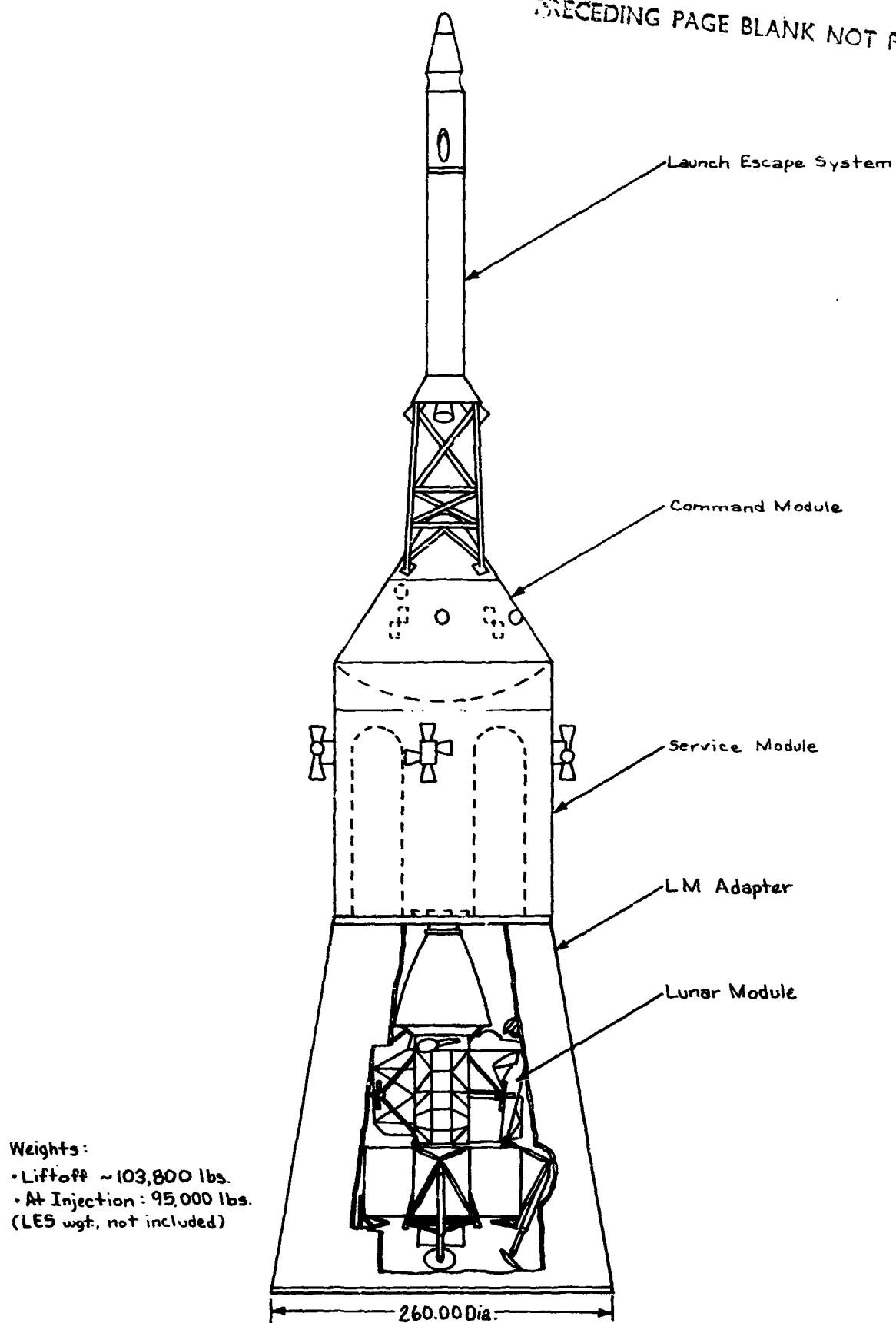


Figure 47

Spacecraft  
Configuration

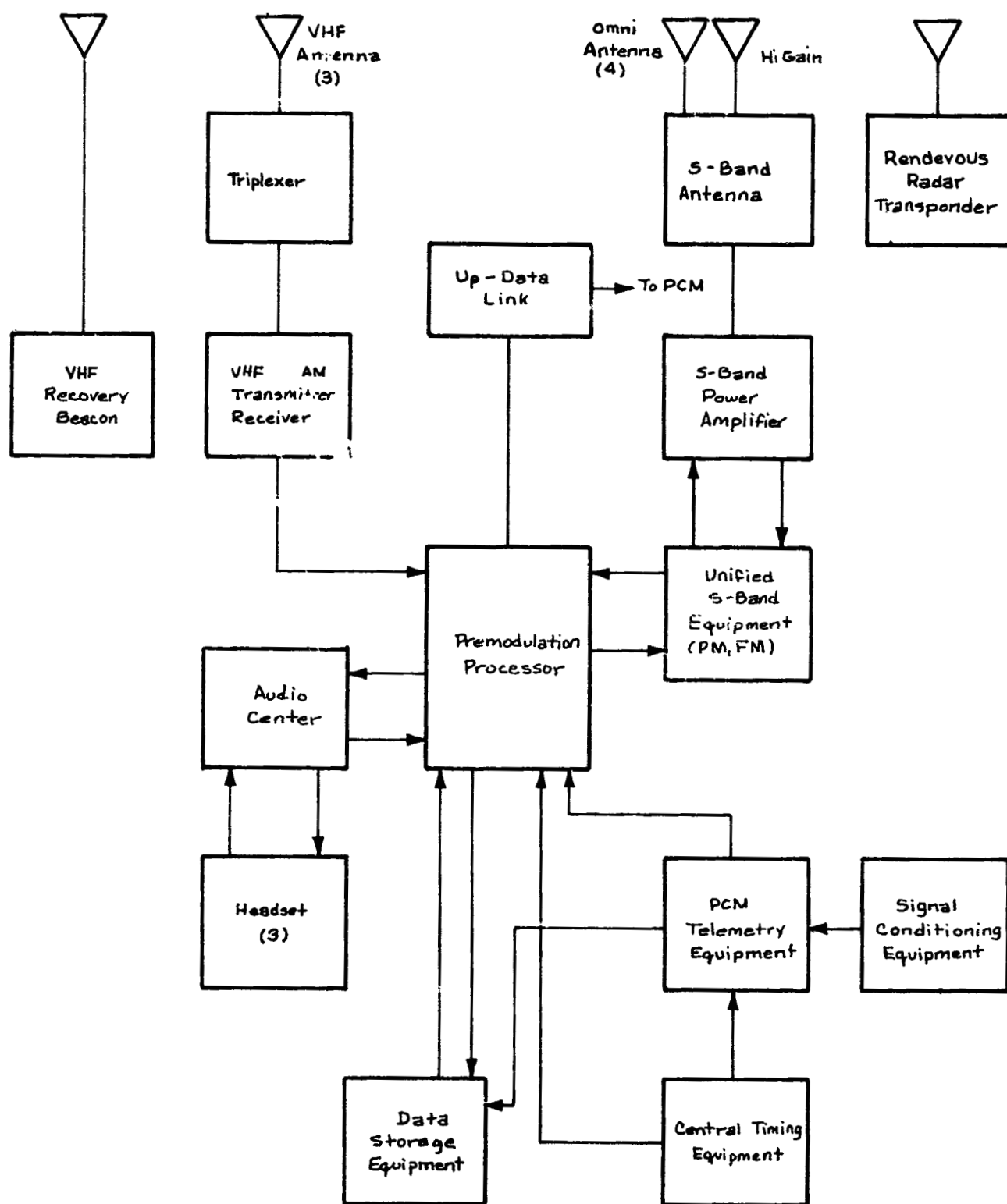


Figure 48

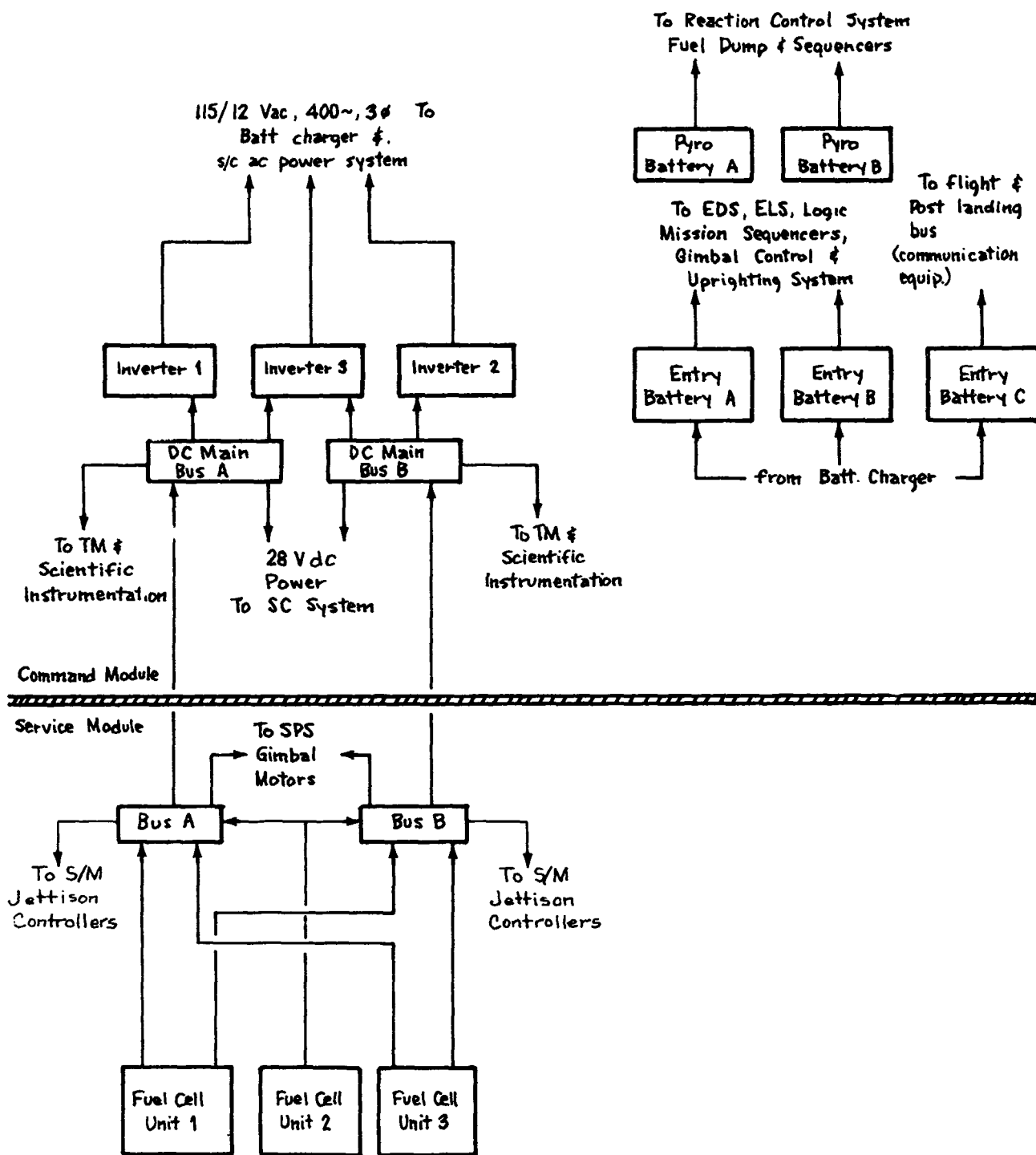


Figure 49

Spacecraft Electrical  
Power & Distribution System

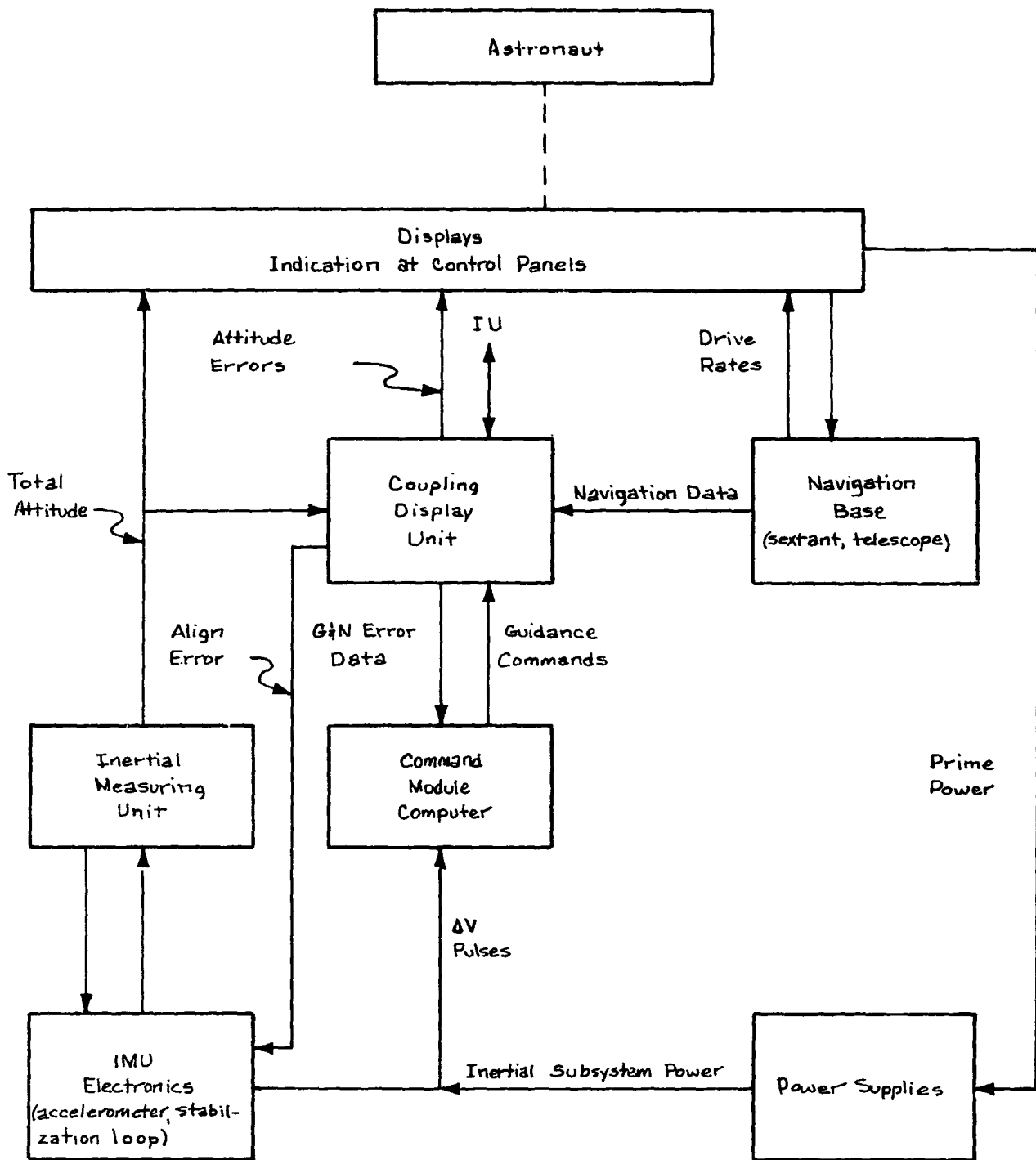


Figure 50

Spacecraft Guidance &  
Navigation System

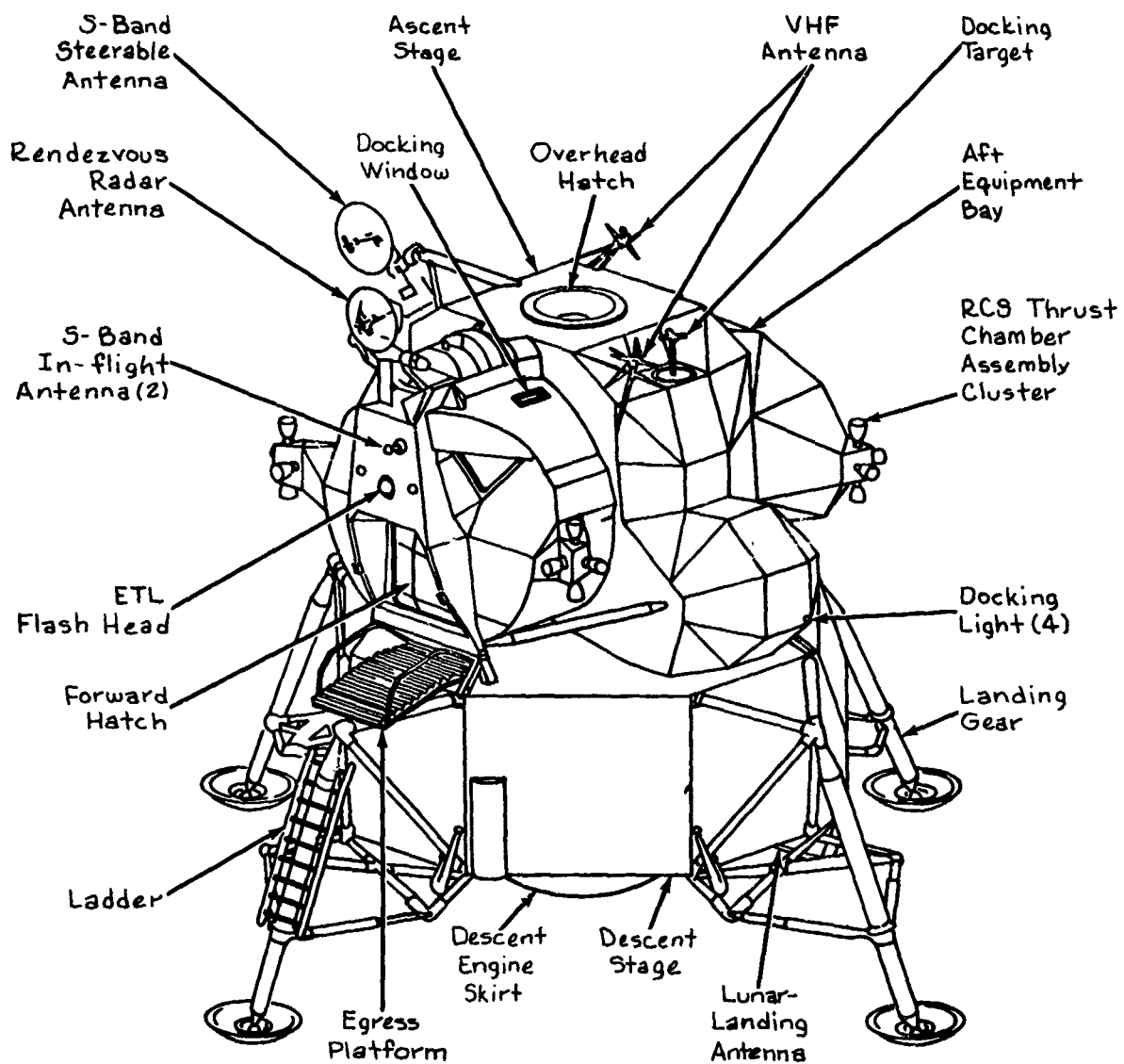


Figure 51

Lunar Module (LM)

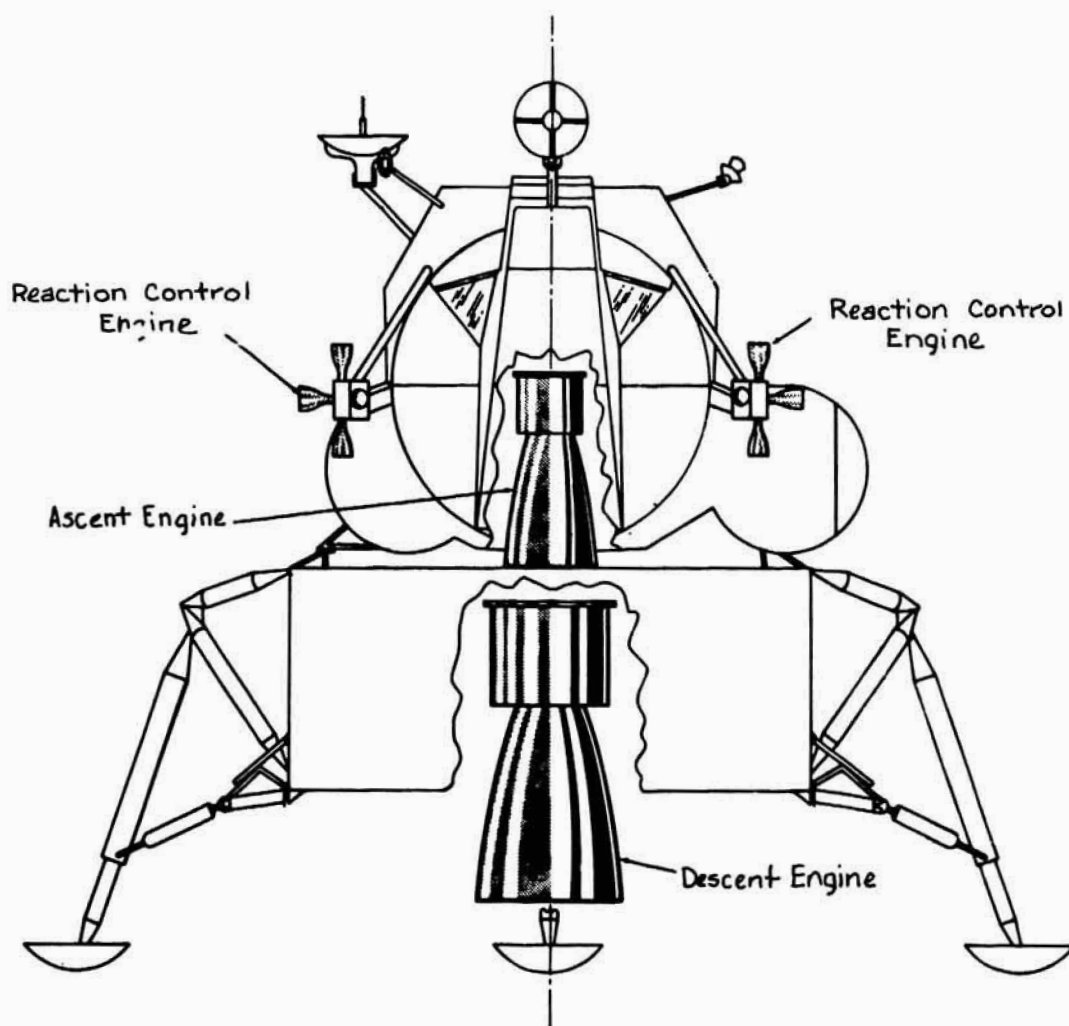


Figure 52

LM Engine Locations



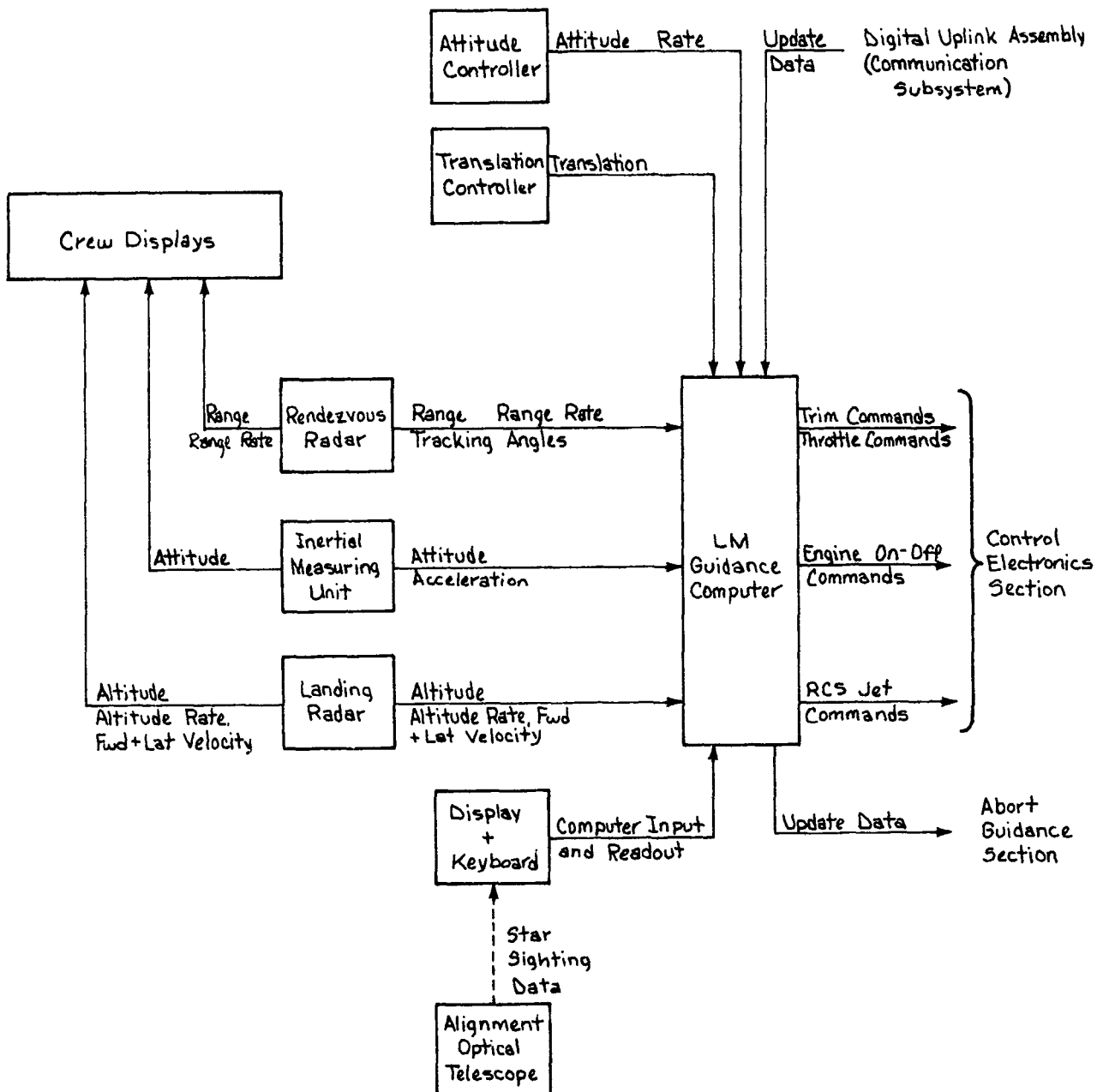


Figure 53

LM Guidance and  
Navigation Section

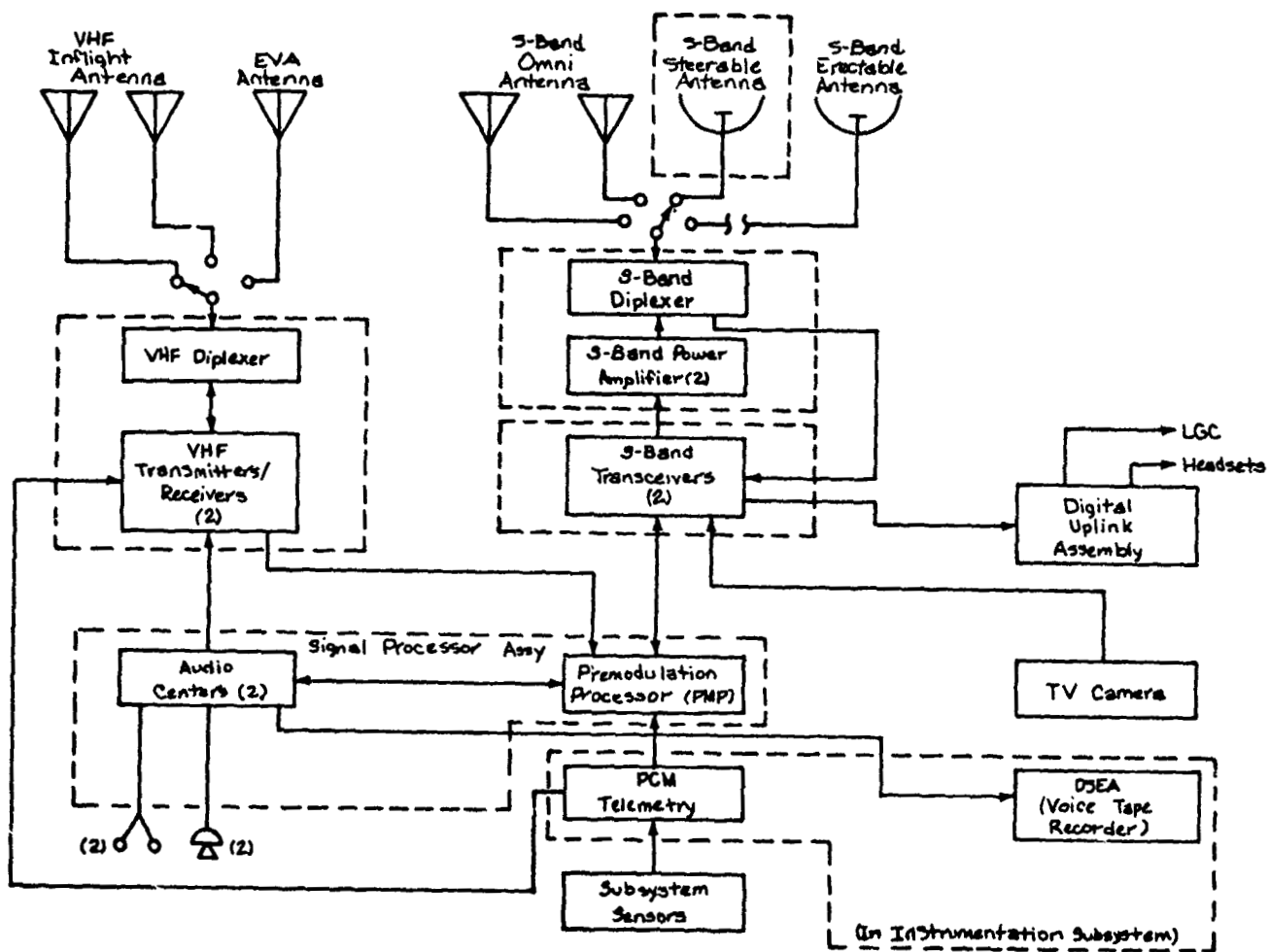


Figure 54

LM Communications Subsystem